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PROPULSION SYSTEMS FOR INITIALLY ACCELERATING A SUPERSONIC-COMBUSTION-RAMJET CRUISING TRANSPORT

by Gerald Knip, Jr., and Leo C. Franciscus

Lewis Research Center
Cleveland, Ohio

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ABSTRACT

A mission study was made to determine which of five propulsion systems would be best for accelerating a hydrogen-fueled, supersonic-combustion-ramjet (SJ) cruise vehicle from takeoff to SJ takeover. Cruise Mach numbers between 7 and 12 were studied. The overall performance of each system was based on range to end of descent. The sensitivity of range to sonic-boom overpressure, vehicle operating weight empty, SJ inlet pressure recovery, SJ exhaust nozzle efficiency, and kinetic effects were determined for one of the propulsion systems.

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PROPULSION SYSTEMS FOR INITIALLY ACCELERATING A SUPERSONIC-COMBUSTION-RAMJET CRUISING TRANSPORT (U)

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SUMMARY

A mission study was made to determine which of five propulsion systems would be best for accelerating a hydrogen-fueled, supersonic-combustion-ramjet (SJ) cruise vehicle from takeoff to SJ takeover. Cruise Mach numbers between 7 and 12 were studied for a vehicle having a takeoff weight of 500 000 pounds (226 800 kg) and a payload of 50 000 pounds (22 680 kg). The five propulsion systems considered were the afterburning turbojet - rocket - supersonic combustion ramjet (TJ-R-SJ), the turboramjet - supersonic combustion ramjet (TRJ-SJ), the afterburning turbojet - dual-mode (subsonic-supersonic combustion) ramjet (TJ-DMRJ), the ejector ramjet - supersonic combustion ramjet (ERJ-SJ), and the ejector dual-mode ramjet (EDMRJ). The overall performance of each system was based on range to end of descent. The results are necessarily preliminary and are subject to change as the propulsion, aerodynamic, and structural elements of a hypersonic airplane evolve.

Of the five engine systems considered, the two most promising were (a) the turboramjet - SJ and (b) the afterburning turbojet - dual-mode ramjet. The SJ of (a) operates from Mach 6.5 to end of cruise. The dual-mode ramjet of (b) operates as a subsonic combustion ramjet from Mach 1, where it augments the turbojet, to Mach 7. Above Mach 7, this engine operates in the supersonic combustion mode. The turbomachinery of both systems operate from takeoff to Mach 3.1.

The sensitivity of range to sonic-boom overpressure, vehicle operating weight empty (OWE), SJ inlet pressure recovery, SJ exhaust nozzle efficiency, and kinetic effects were determined for the turboramjet-SJ propulsion system. With the nominal values of these parameters, the range to end of descent was 7140 nautical miles (2176 m) for a cruise Mach number of 10. The sonic-boom overpressure during Mach 10 cruise was 0.5 pound per square foot (23.9 N/m^2). Decreasing the nominal value of the transonic sonic-boom overpressure limit of 2.5 pounds per square foot (119.7 N/m^2) by 5 percent results in a 1-percent decrease in range. Sonic-boom levels below 2.2 pounds per square foot (105.3 N/m^2) were unattainable. A 5-percent decrease in the OWE results in an 11-percent increase in range. A 25-percent reduction in the nominal pressure recovery schedule results in a 9.6-percent decrease in range. Decreasing the nominal exhaust nozzle efficieny by 4 percent results in a 5.8-percent decrease in range. Accounting for kinetic effects in the exhaust nozzle results in a 5.5-percent decrease in range.

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INTRODUCTION

In recent years there has been renewed interest in airbreathing propulsion systems, both in the supersonic and hypersonic speed ranges. As applied to transport vehicles, the supersonic (SST) entry planned by the United States is being designed for a cruise Mach number of 2.7 and a range of 3500 nautical miles (6.48×10^6 m). Future transport vehicles will be required to have longer ranges, and a lower sonic-boom level would be desirable.

Traffic projections (refs. 1 and 2) show a need for future transports capable of flight ranges of 5000 to 8000 nautical miles (9.3×10^6 to 14.8×10^6 m). To prevent trip times from becoming excessive, a cruise speed greater than Mach 2.7 is desirable.

For a range of 8000 nautical miles (14.8×10^6 m), figure 1 indicates a total trip time of approximately 16 hours (including 2 hr for ground transportation, holds, etc.) for a subsonic jet. An average acceleration of 0.2 g was assumed in calculating the time for the climb and descent phases of the flight. The total trip time can be reduced to 7.2 hours by increasing the cruise Mach number to 2.7 (SST) or to 4.9 hours by further increasing the cruise Mach number to 5 (HST, hypersonic transport). Thus, increasing the cruise Mach number from 0.9 to 5 results in a very significant reduction in total trip time (65 percent).

Studies, such as in references 3 to 6, have been made for Mach 5 and 6 hypersonic transports. These vehicles attained ranges of about 5000 nautical miles (9.3×10^6 m). To achieve this range at this speed, a fuel having a high energy content and a high heat sink capacity is required (ref. 4). Therefore, hydrogen is the primary fuel candidate for a hypersonic vehicle. Propulsion systems being considered for this vehicle are the turboramjet and the turbofan ramjet. These engines, like those proposed for the SST, use subsonic combustion.

Cruise speeds beyond Mach 5 may also be considered for future generation transports. If the cruise speed is increased from Mach 5 to Mach 10, for example, the total trip time for an 8000 nautical mile (14.8×10^6 m) trip is reduced from 4.9 to 3.75 hours. Although this is only a 23-percent reduction in terms of total trip time, it is a substantial reduction in terms of flight time (40 percent).

For cruise speeds beyond Mach 5 or 6, the hydrogen burning ramjet using supersonic combustion (SJ) appears to offer an attractive range capability (ref. 5). The SJ, like the subsonic combustion ramjet, does not have a static thrust capability. Therefore, another propulsion system is required for the initial acceleration phase of the flight (takeoff to SJ takeover).

For this phase, there are many engines from which to choose. In making a selection, the tradeoff involved is one of engine weight and acceleration fuel or propellant. In terms of the mission, the acceleration engine is important because of the fact that its weight and the acceleration fuel weight represent a significant part of the vehicle gross weight. The engines selected for this study (see fig. 2) cover the spectrum from the lightweight but low-specific-impulse rocket to the heavy but high-specific-impulse turboramjet. These engines also represent various levels of technology.

The purpose of the present study was to evaluate various propulsion systems comprised of these initial acceleration engines in combination with the supersonic burning ramjet. Range to end of descent was used as the measure of performance. The modes of propulsion considered were

System 1: Afterburning turbojet - rocket - supersonic combustion ramjet (TJ-R-SJ)
System 2: Turboramjet - supersonic combustion ramjet (TRJ-SJ)
System 3: Afterburning turbojet - dual-mode (subsonic-supersonic combustion) ramjet (TJ-DMRJ)
System 4: Ejector ramjet - supersonic combustion ramjet (ERJ-SJ), and
System 5: Ejector dual-mode ramjet (EDMRJ)

Each of the five systems consists of from one to three engines including the supersonic burning ramjet. System 1 uses three separate engines; however, two of the three engines use basically present state-of-the-art technology. Systems 2, 3, and 4 each use two separate engines, while system 5 consists of only one engine. The variation in engine thrust-to-weight ratio and specific impulse associated with each system influences the aircraft in terms of size, structural weight, and fuel weight. A mission study was thus performed for a general transport configuration to determine the most promising system in terms of range. Cruise Mach numbers between 7 and 12 were considered. Figure 3 indicates the basic vehicle configuration used in the study. The vehicle has a TOGW of 500 000 pounds (226 800 kg) and a payload (250 passengers plus baggage) of 50 000 pounds (22 680 kg). While a much more sophisticated configuration may be required to attain the aerodynamic performance level used in the study, this simplified configuration permitted airframe aerodynamics and weight estimates to be readily calculated when changes in the airframe resulted from perturbations in the propulsion system.

At hypersonic speeds the descent phase of the flight can represent a significant portion of the vehicle's total range. This is due to both the total energy which the vehicle possesses and the effect of centrifugal lift. A gliding descent was thus considered to determine its effect on the optimum cruise speed and range.

Subsonic loiter times at the end of the descent were calculated for the TRJ-SJ and EDMRJ systems. The purpose of calculating these times was to determine what, if any, additional advantage could be obtained from the turbomachinery-type engine.

A sensitivity study was also made using the TRJ-SJ system to determine the effect of sonic-boom overpressure, vehicle operating weight empty, SJ inlet and nozzle efficiency, chemical nonequilibrium expansion in the exhaust nozzle, and cruise cooling requirements on range.

SYMBOLS

A_e	operating engine capture area downstream of wing shock, ft^2; m^2
A_t	rocket throat area, ft^2; m^2
C_f	friction coefficient
C_L	lift coefficient
C_T	thrust coefficient, F_n/q_oA_e
D	vehicle drag, lb; N
d	maximum fuselage diameter, ft; m
F	thrust, lb; N
f/a	fuel-air ratio
H_v	fuel heating value, Btu/lb; J/kg
ΔH	fuel heat sink capacity, Btu/lb; J/kg
h	enthalpy, Btu/lb; J/kg
I	fuel or propellant specific impulse, sec
K_N	supersonic-combustion-ramjet exhaust nozzle efficiency (see p. 11)
L	vehicle lift, lb; N
l	fuselage length, ft; m
M	Mach number
q	dynamic pressure, ρV^2, lb/ft^2; N/m^2
R_e	Reynolds number, $\rho V^2 1/\mu$
S	wing platform area, ft^2; m^2
v	velocity, ft/sec; m/sec
W	vehicle weight, lb; kg
$\dot{w}$	flow rate, lb/sec; kg/sec
α	vehicle angle of attack, deg
φ	equivalence ratio, (f/a)/(f/a)s

μ dynamic viscosity, slugs/ft-sec; $N\text{-sec/m}^2$

ρ density, slugs/ft^3; kg/m^3

η_e overall engine efficiency, IV/H_v

Subscripts:

a air

f fuel

G gross

n net

p propellant

r subsonic combustion ramjet

s stoichiometric

v vacuum

ANALYSIS

As a means of evaluating each of the five propulsion systems considered herein for use with a hypersonic transport, a mission study was performed for cruise Mach numbers between 7 and 12. The performance of each system was based on range to end of descent.

Vehicle

Figure 3 indicates the basic vehicle configuration used for the mission study. The vehicle consists of a delta-wing body configuration having a takeoff gross weight (TOGW) of 500 000 pounds (226 800 kg) and a payload of 50 000 pounds (22 680 kg) (250 passengers plus baggage).

The hypersonic transport configuration parameters are as follows:

Wing area (gross), ft^2 (m^2)	9000(836)
Thickness ratio	0.03
Aspect ratio	1.5
Leading edge sweep, deg	69
Leading edge radius, in. (m)	3(0.076)
Tail area (gross), ft^2 (m^2)	1000(92.9)
Thickness ratio	0.03
Aspect ratio	2
Leading edge sweep, deg	63
Leading edge radius, in. (m)	2(0.051)
Fuselage	
Overall fineness ratio	15.2
Nose fineness ratio	4

Wing. - Based on preliminary mission studies, a leading edge sweep angle of 69^{o} was selected for the delta wing. For a fixed wing area, the wing weight decreases with increasing sweep angle because of the reduced bending loads. However, the lift-curve slope ($\partial C_L/\partial\alpha$) also decreases. High sweep angles also alleviate the structural heating problem and minimize the drag associated with the blunted leading edge. A blunt rather than a sharp leading edge is required to withstand the stagnation temperatures at hypersonic speeds. For the present study, a representative leading edge radius of 3 inches (0.076 m) was used.

The takeoff wing loading (W/S) was fixed at a relatively low value of 56 pound per square foot (273 kg/m^2) to achieve an acceptable takeoff speed (160 n mi/hr (82.4 m/sec), $\alpha_{max} = 15^{o}$) without the aid of high lift devices. The maximum angle of attack at takeoff is limited by the ground clearance of the vehicle's afterbody. Based on the results of reference 6 a wing thickness chord ratio of 0.03 was used.

Fuselage. - From preliminary studies, a fuselage fineness ratio (l/d) of 15.2 was selected. The fuselage consists of a conical forebody having a fineness ratio of 4 and a cylindrical afterbody. The fuselage contains all of the fuel, liquid oxygen (if required), passengers, and the initial acceleration engines. These engines were located in the base of the fuselage. Requirements as to stability and control were not considered. The two-dimensional variable geometry inlet for the initial acceleration engines and the podded SJ's were located beneath the wing to take advantage of the wing precompression and to minimize the effect of the angle of attack on the airflow entering the engine. Air for the initial acceleration engines is ducted from the inlet through the fuselage to the engines. During SJ operation this air supply is shut off. A detailed study of this operation was not made. For the present study a drag penalty was included based on using the variable ramp inlet at an angle of 6°.

Flight Path

Based on previous studies, a representative flight path was selected. This path was then subjected to various constraints. The equations of motion were then integrated along the resultant flight path to obtain a time, range, and weight history of the flight from takeoff to end of descent.

Sonic boom. - The first constraint encountered along the flight path after takeoff was sonic-boom overpressure (ref. 7). For this study, the ground overpressure was limited to 2.5 pounds per square foot (119.7 N/m^2). To satisfy this constraint, the altitude of the flight path was adjusted between Mach 1 and 2.75. To be conservative, the boom characteristics for the present configurations were based on those of a JP fueled SST delta wing configuration (SCAT-17) since it represents a vehicle which does not have optimum boom characteristics. The unrestricted flight path was then followed from Mach 2.75 to approximately 4.5 at which point the next constraint is encountered.

Engine pressure. - For this study, the engine duct pressure was arbitrarily limited to a representative value of 200 psi (1.379×10^6 N/m^2). For each Mach number, the altitude corresponding to this pressure was determined by using the inlet pressure recovery schedule given by military (Mil E-5008B) (ref. 8) and the vehicle angle of attack. The engine pressure limit thus determines the flight path between Mach 4.5 and about 6.5. The magnitude of this pressure affects both the engine's weight and its performance. High duct pressures result in not only improved engine performance but also increased engine weight. Because supersonic combustion ramjets are used at the higher Mach numbers, the engine pressure constraint is bypassed. This leads to the last constraint, that of aerodynamic heating which determines the flight path above Mach numbers of about 7.

Aerodynamic heating. - The temperatures attained by the various surfaces of a vehicle are a function of the emissivity of the surface, the vehicle angle of attack, Mach number, and altitude. For applications up to about 1500° F (1089° K), superalloys appear to be satisfactory. A typical radiation equilibrium temperature obtained on the lower surface of a flat plate is indicated in figure 4. This temperature occurs at a location 1 foot (0.3048 m) aft of the leading edge assuming an emissivity of 0.9, an angle of attack of 6°, and a turbulent boundary layer. For the current study, the flight path was increased arbitrarily so as not to exceed metal temperature limitations. The maximum temperature occurs at the cruise condition. The maximum cruise Mach number and altitude considered were Mach 12 and 140 000 feet (42 672 m).

In comparison to the SST which will cruise at an altitude of about 70 000 feet (21 336 m), the HST will probably cruise at an altitude of 100 000 feet (30 480 m) or more. According to reference 9, the hazards from radiation at these altitudes are less than at 70 000 feet (21 336 m). Therefore, radiation exposure to passengers of future hypersonic transports is not expected to be any greater than that predicted for the SST.

Cruise and descent. - After achieving the desired cruise Mach number, the initial cruise altitude was adjusted to maximize the Breguet factor (IVL/D). At this altitude the supersonic combustion ramjets were throttled back (equivalence ratio, φ, <1) for cruise. To maximize the cruise range, the Breguet factor was held constant by adjusting the vehicle's altitude. The vehicle followed a Breguet flight path until 10 percent of the hydrogen fuel load remained (landing maneuver and reserve). No hydrogen fuel boiloff was considered. In lieu of a more detailed study, a gliding descent was assumed for the descent portion of the flight. The SJ inlets were blocked off during this phase of the flight to relieve the engine cooling requirements (no cooling assumed).

Airframe Structure

Because of the high skin temperatures (discussed previously) and the liquid temperature of hydrogen (36° R (20° K)), hypersonic airplanes pose difficult structural and material problems. For transport vehicles, radiative rather than absorptive systems are being considered. This is due to both the time per flight a transport vehicle is exposed to this thermal environment and the number of reuses required of it. Both hot and cold radiative structures and integral and nonintegral fuel tanks are being considered. To withstand these temperature extremes, both superalloys (RENE-41 and Hastalloy) and refractory alloys (columbium and molybdenum) will be required for fabricating the various components.

For the present study, a cold structure radiative system was used. The primary elements are indicated at the top of figure 5. These elements are the superalloy skin, the insulation, and the hydrogen tanks which serve as an integral part of the load bearing structure.

Typical unit weights for the fuselage and the wing are indicated at the bottom of figure 5. The fuselage unit weight for a typical vehicle was 5.38 pounds per square foot (26.2 kg/m^2) of fuselage surface area while the unit weight for the wing was 6.6 pounds per square foot (32.2 kg/m^2) of gross wing planform area. The base area of the fuselage was not insulated. For the wing, only the exposed lower surface was insulated. The upper surface may require some insulation depending on the load bearing material used. The structural weights for the load bearing structure are based on the vehicle's geometry, flight path, and the empirical equations of reference 10. Weights for the heat protection system are based on the temperature environment during the cruise portion of the mission and reference 11.

Airframe Aerodynamics

The lift and drag for the given vehicle configuration and trajectory were determined by totaling the values of the contributing components (wing, fuselage, podded nacelles, and vertical tails). Interference effects were included as given in reference 12. The lift-curve slope was determined by the method of reference 13. Figure 6 indicates the lift-curve slope and the zero-lift drag coefficient as a function of Mach number for a typical vehicle.

The zero-lift drag coefficient includes the wave and friction drags of the various vehicle components and the cold flow drag of the SJ (started at Mach 6.5). The wave drag coefficients for the wing, fuselage, nacelles, and vertical tails were determined from correlation of data, and from linearized and Newtonian theory for sharp leading edge wings and forebodies (refs. 14 to 16). Blunt leading edge drag for the wing, inlet, vertical tails, and nacelles were included. An incompressible value of the skin friction coefficient for a flat-plate turbulent boundary layer was calculated by the Prandtl-Shlichting equation:

$$C_f = \frac{0.46}{(\log_{10}Re)^{2.58}}$$

This value was corrected for compressibility effects using the reference enthalpy method. The cold flow drag for the SJ's included the additive and bypass drags. The additive drag was based on the capture mass flow ratio and a 6° half-angle cone (ref. 17). The bypass drag was based on a 0.9 normal shock pressure recovery and a sonic nozzle. The drag due to lift (or the induced drag) was calculated from the vehicle's normal force coefficient and the angle of attack.

Engine Cycles

In the present study, five propulsion systems were considered for propelling a hypersonic cruise transport from takeoff to landing. The systems considered were as follows:

System 1: Afterburning turbojet - Rocket - supersonic combustion ramjet
(TJ-R-SJ)

System 2: Turboramjet - supersonic combustion ramjet (TRJ-SJ)

System 3: Afterburning turbojet - Dual-Mode Ramjet (TJ-DMRJ)

System 4: Ejector ramjet - supersonic combustion ramjet (ERJ-SJ)

System 5: Ejector Dual-Mode Ramjet (EDMRJ)

System 1 consists of three separate engines while systems 2 to 4 each consist of two engines. For system 5, the SJ is combined with the ERJ to form a single engine known as the ejector dual-mode ramjet. For all five systems, the SJ is used for the final acceleration and cruise phases of the flight. The other engines within each system accelerate the vehicle from takeoff to SJ takeover. Each of the engines is discussed in this section. Since the SJ is used in all five systems, it is discussed first and in greater detail.

Supersonic combustion ramjet (SJ). - The SJ is similar to the subsonic combustion ramjet except that the flow velocity remains supersonic throughout the engine; since less diffusion is required, the static temperatures and pressures throughout the engine are lower. Structural and heating loads are thus reduced. The inlet pressure recovery is higher and the exhaust nozzle dissociation losses are lower due to the lower combustion temperature. For the present study, the SJ inlet total pressure recovery, figure 7, was obtained from reference 6. The experimental data of reference 18 are also shown. These data represent the highest point value, not a mass average value of total pressure measured in the inlet throat. Thus the pressure recovery schedule used in the present study represents a goal for an SJ inlet. The pressure recovery curve for the initial acceleration engine in figure 7 will be discussed later. At Mach 9, the inlet pressure recovery for a subsonic combustion ramjet is seen from the initial acceleration engine curve to be about 0.1. For an equivalence ratio of 1.0, the calculated combustion temperature would be about 6500° R (3611° K). Compared to these values at the same flight Mach number, the SJ inlet pressure recovery is seen to be much higher (about 0.6), and the combustion temperature based on a 90-percent combustion efficiency is calculated to be about 5000° R (2778° K).

Although the inlet and exhaust nozzle losses are lower for the SJ, the combustion losses will be higher due to the higher momentum loss incurred by heat addition at supersonic velocities. Mixing, shock, and friction losses in the combustor will also be higher. Based on analytical engine data, the performance of the SJ appears to be superior to that of the subsonic burning ramjet above a Mach number of about 6 (ref. 19). The performance used in the present study considered real gas effects and chemical equilibrium expansion in the exhaust nozzle.

A study was made of the nozzle dissociation or kinetic effects (as it affects range) using Bray's method as applied in reference 20. The combustion efficiency was assumed to be 0.9. Fuel was injected sonically and parallel with the air stream at a temperature of 2000° R (1110° K). Based on reference 6, an exhaust nozzle process efficiency (K_N) of 0.928 was assumed. This efficiency is defined as

$$K_N = \frac{(h_3 - h_e)}{(h_3 - h_e)_{IDEAL}}$$

where the subscripts 3 and e represent the exhaust nozzle entrance and exit, respectively.

In terms of geometry, the engine has an overall area ratio (nozzle exit area to inlet captive area) of 1.0. In the present study, a combustor area ratio of 1.35 was chosen since it appeared to be the best compromise between overall engine performance and thermal choking.

Thermal choking in the engine is affected by flight Mach number, inlet contraction ratio, combustor area ratio, and fuel-air ratio. In terms of engine performance, a high inlet contraction ratio and a low combustor area ratio result in the best specific impulse. To avoid thermal choking in a fixed geometry SJ at low flight Mach numbers, especially below Mach 7, the combustor can be designed for either (1) the upper Mach number range (area ratio near 1.0) and operated at fuel-air ratios below stoichiometric (equivalence ratio less than 1.0), (2) the lower Mach number range (area ratio greater than 1.0) and operated stoichiometrically, or (3) with distributed fuel injection points to yield an effective variable-area combustor. Because the present engine operates principally in the upper Mach number range, Mach 6 to 12, it was designed according to point (1).

However, a detailed study of this engine performance tradeoff was not made. Thus in the present study, the allowable equivalence ratio was dictated by the inlet contraction ratio since the combustor area ratio was fixed at a value of 1.35. Figure 8 indicates, for various flight speeds and inlet contraction ratios, the maximum equivalence ratio at which the engine can be operated without encountering thermal choking. Below about Mach 7, all inlet contraction ratios require the engine to be operated at φ's less than 1.0. Below Mach 5, φ must be less than 0.5. For a contraction ratio of 10, the equivalence ratio varied from about 1.0 at Mach 7 to about 0.3 at Mach 5. As the contraction ratio is increased, these values of φ must be decreased. These restrictions influence the resultant engine performance obtained for a given inlet contraction ratio.

Figure 9 indicates, for three inlet contraction ratios, the variation in specific impulse and thrust coefficient with Mach number. Included in this performance is the inlet additive drag based on an inlet having a design Mach number of 8. For the present study the design Mach number was assumed to be independent of cruise Mach number. No specific inlet performance penalty was considered for cruise Mach number greater than 8. The additive drag is based on a 6^o half-angle cone and the calculated capture mass flow ratio. Above about Mach 7.5, both the specific impulse and the thrust coefficient ($C_T = F_n/q_0A_e$) increase with inlet contraction ratio. Below Mach 7.5, the best specific impulse was obtained with a contraction ratio of 15. The thrust coefficient decreases rapidly with increasing contraction ratio. For example at Mach 5, the thrust coefficient is 0.6 for a contraction ratio of 10 as compared to 0.13 for a contraction ratio of 25. The primary reason for this decrease in the thrust coefficient is the lean fuel-air ratio at which the engine is forced to operate to avoid thermal choking (discussed previously). A secondary reason is the shock spillage below Mach 8 and its associated additive drag. Thus, the contraction ratio used will depend to a large degree on the engine used for the initial acceleration phase of the flight. The inlet contraction ratio and the SJ takeover Mach number were, therefore, variables to be determined in the study for each of the five propulsion systems.

A problem which must be considered in using the SJ as a cruise powerplant is engine cooling. It is important that the equivalence ratio required to cool the engine during cruise be less than the equivalence ratio required for thrust. Cooling requirements are affected by engine size or capture area, maximum allowable structural temperature, engine geometry, and Mach number.

In the present study, all internal surfaces except the inlet leading edge were regeneratively cooled with hydrogen. The leading edge was assumed to be constructed of a refractory metal. Gas-side heat-transfer rates for the internal surfaces of the engine were determined by the method of references 21 and 22. The hydrogen before cooling was assumed to be a gas at 100° R (56° K). The equivalence ratio required to maintain a constant internal wall temperature during cruise (in real engine all parts of the engine are not at the same temperature) was determined from the following simplified expression:

$$\dot{w}_f = \frac{\text{Total cooling load}}{\eta_e \, \Delta H} \; \frac{\text{lb fuel}}{\text{sec}} \left(\frac{\text{kg}}{\text{sec}}\right)$$

$$\varphi_c = \frac{\dot{w}_f/\dot{w}_a}{0.0292}$$

The η_e represents an "effective" or an overall heat-transfer efficiency between the hot wall and the hydrogen. A value of 0.85 was used in the present study. Cooling requirements for the cutaway or scalloped type SJ engine used in the study are shown in figure 10. For an inlet contraction ratio of 10 and a cruise Mach number of 10, the φ required for cooling decreased from 1.0 to 0.56 as the allowable structural or internal wall temperature was increased from 500° to 2000° R (277° to 1110° K). Supersonic combustion engine configurations having more wetted surface area may require values of φ_c higher than these.

The effect of cruise Mach number on φ_c is indicated in figure 11. The φ required to maintain a constant internal wall temperature of 2000° R (1110° K) ranged from 0.24 at Mach 8 to 0.6 at Mach 12. These requirements increase with inlet contraction ratio due to the increase in the static temperature and pressure of the incoming air. For a contraction ratio of 25, φ_c ranged from 0.3 at Mach 8 to 0.63 at Mach 12. As indicated by the thrust requirement curve, the φ's required for cruise thrust are adequate for handling the engine cooling requirements. The thrust requirement curve drops at Mach 12 because the engine capture area required to accelerate the vehicle to Mach 12 is greater than that required for cruise.

Up to this point, the performance and cooling requirements of the present SJ have been discussed. Another item of importance in a mission study is engine weight. The installed weight of the podded scalloped type SJ engine was based on a value of 76.6 pounds per square foot (374 kg/m^2) of capture area. This value was obtained from references 6 and 23.

Afterburning turbojet (TJ). - The characteristics of the TJ used in the study are as follows:

(1) Sea level static compressor pressure ratio, 8
(2) Turbine inlet temperature, 3660° R (2033° K)
(3) Primary combustor efficiency, 0.98
(4) Afterburner efficiency, 0.93
(5) Overall fuel-air ratio during acceleration, 0.0292 (stoichiometeric)
(6) Sea level static thrust, 44 000 pounds (195 900 N) for a corrected airflow of 340 pounds per second (154.2 kg/sec)

The engine performance was based on full expansion to ambient conditions and corrected for nozzle performance based on an inverted plug (expansion-deflection) nozzle. A two-dimensional variable geometry inlet located in the wing pressure field captured the entire free-stream tube at Mach 3.1 (operating limit for TJ). Below Mach 3.1, a spillage drag was accounted for in the engine performance. The additive drag was based on a 6° wedge (ref. 24) and the engine capture mass flow ratio. Typical values of the nozzle thrust coefficient and the additive drag coefficient are given in table I. The inlet pressure recovery schedule used in the study is shown in figure 7. Above Mach 1.7, the recovery schedule for the initial acceleration engines corresponds to military (Mil-E-5008B) (ref. 9). Below Mach 1.7, a value of 0.95 was arbitrarily used to account for diffuser losses.

Based on installed engine weights for the SST and the empirical equations of reference 10, a value of 23.8 pounds (10.8 kg) of engine per pound (0.4536 kg) of sea level static airflow was used in calculating the weight of the installed TJ. This weight includes (1) inlet and ducting, (2) basic engine, (3) plumbing, and (4) nozzle.

TABLE I. - TYPICAL INLET AND

NOZZLE PERFORMANCE

Mach number	Additive drag coefficient (a)	Nozzle thrust coefficient (b)
0.6	0.051	0.889
1.0	.135	.922
1.5	.168	.954
3.1	0	.981

[a]Based on inlet capture area.
[b]Ratio of actual nozzle thrust to ideal thrust for full expansion to ambient pressure.

Turboramjet (TRJ). - The inline TRJ operates essentially as a TJ from takeoff to Mach 3.1. Therefore, the same engine characteristics as for the previously described TJ were used. From Mach 3.1 to SJ takeover, the inlet air is bypassed around the turbomachinery and ducted to the afterburner. The engine then operates as a subsonic combustion ramjet. The ramjet was arbitrarily sized so that the ratio of the ramjet corrected airflow at Mach 3.1 to the sea level static airflow of the TJ was 0.5. The engine performance was again based on full expansion to ambient conditions and corrected for nozzle performance based on an inverted plug. The inlet performance was handled in a manner similar to that for the TJ except that the inlet design point was selected at Mach 4.5 (ref. 24). Therefore, from Mach 4.5 to engine shutdown or SJ takeover, the inlet additive drag is zero since the inlet captures a full stream tube.

The installed weight of the turboramjet is shown in figure 12 as a function of engine sea level static airflow for three maximum operating or cutoff Mach numbers. The installed weight was determined by use of the empirical equations of reference 10 and the installed engine weight data of reference 25. For a cutoff Mach number of 6.5 and a sea level static airflow of 368 pounds per second (167 kg/sec), the installed engine thrust to weight ratio is 4.

Rocket (R). - In determining the optimum (based on range) liquid oxygen - hydrogen rocket for the TJ-R-SJ propulsion system, various rocket motor parameters were investigated. Initially, the throat area and the design pressure ratio were fixed at 1.6 square feet (0.149 m^2) and 666, respectively. Ranges of chamber pressure (600 to 1000 psi (4.1 to 6.9×10^6 N/m^2)), equivalence ratio (0.75 to 1.2), and shutdown speed (Mach 4 to 5) were then investigated to determine the best value of each based on range for a cruise Mach number of 8. With the best value of these parameters, the throat area (1.2 to 1.6 ft^2 (0.11 to 0.15 m^2)) and the design pressure ratio (263 to 666) were varied in turn. In this study, the rocket performance was based on the equilibrium data of reference 26. The resultant characteristics of the rocket used in the study are as follows:

Throat area, 1.6 square feet (0.149 m^2)

Chamber pressure, 1000 psia (6.9×10^6 N/m^2)

Equivalence ratio, 1.0

Design pressure ratio, 666

Characteristic velocity, 6900 feet per second (2103 m/sec)

Thurst correction factor, 1.0

Specific impulse, 421 seconds

The weight of the rocket W_{ROC} was based on an empirical equation from reference 10:

$$W_{ROC} = 250 + 0.0033\, F_{VAC}\left(\frac{A_e}{A_t}\right)^{0.196}$$

Dual-mode ramjet. - The dual-mode ramjet considered in this study is a fixed geometry engine which operates in the subsonic combustion mode between Mach numbers of 3 to 7 and then as a supersonic combustion ramjet for the final acceleration and cruise part of the flight.

The inlet has swept or scalloped leading edges for efficient air spillage at low Mach numbers and reducing leading edge heating at high Mach numbers. The combustor is a diverging duct with a cross-sectional area ratio of 1.35. Fuel injectors are located in forward and aft positions in the combustor. Unlike other subsonic burning engines, the entire nozzle is a diverging duct and has no convergent section and physical throat.

For the subsonic combustion mode, air is decelerated through a normal shock in the forward part of the combustor to some subsonic Mach number. Fuel is injected at the aft injectors. The heat addition then accelerates the gases to sonic velocity, thus achieving a thermal throat in the nozzle.

During supersonic combustion, air is decelerated to a supersonic Mach number at the combustor entrance and fuel is injected through the forward injectors. Combustion takes place and supersonic velocities are maintained at the combustor exit. The gases are then expanded through the nozzle.

The inlet is designed to capture a full stream tube of air at Mach 8 and above. Because of the complicated flow fields for an inlet with a scalloped leading edge, any analytical method of determining the off-design operating characteristics would be extremely difficult and time consuming. Therefore, the inlet drag and spillage characteristics of a 6^o conical inlet were used. Based on the experimental data of reference 18, this assumption is not unreasonable. The inlet total pressure recovery schedule was obtained from reference 6 (see fig. 7).

Based on preliminary analysis, the inlet contraction ratio that optimized range is 10. Below Mach 4, the inlet unstarts with this contraction ratio. Therefore, the excess flow is bypassed through bypass doors. The problem could also be averted by decreasing the contraction ratio. However, this would require some type of movable structure to increase the throat area to accommodate the mass flow. The inlet operating characteristics for the fixed geometry inlet with bypass doors and those for a variable geometry inlet are compared in figure 13. The drag coefficients for the fixed inlet are seen to be about 6 to 8 times greater than those of the variable inlet between Mach numbers of 1 to 3. At Mach 3.1, the fixed inlet captures about 35 percent of the available mass flow; 30 percent is spilled, and 35 percent is bypassed. For the variable inlet at Mach 3.1, 70 percent of the available mass flow is captured and 30 percent is spilled. Therefore, the thrust coefficients for an engine with a variable inlet might be expected to be twice those of the fixed inlet engine at this Mach number.

If the ramjets are considered for augmenting the turbojets at transonic speeds, the effect of variable geometry becomes even more important. As seen in figure 13, the fixed inlet captures only 10 percent of the available mass flow transonically, whereas the variable inlet captures 60 percent. This factor affects not only the ramjet's thrust, but also the sizing of the TJ, the overall propulsion weight, the acceleration fuel requirements, and thus the vehicle's range. Since the effect on range may be appreciable, it was considered in the present study.

The installed weight of the fixed geometry engine was based on a value of 107 pounds per square foot (523 kg/m^2) of capture area (ref. 23), while the weight of the movable geometry engine was treated as a variable.

Ejector ramjet (ERJ). - The ejector ramjet engine powers the vehicle from takeoff to SJ takeover. This engine is similar to a basic subsonic burning ramjet. The engine has a conventional inlet and diffuser. Located at the end of the diffuser is a cluster of rocket chambers followed by a mixer-diffuser section in which the rocket exhaust gases are mixed with the incoming air. Following the mixer diffuser are an afterburner and a nozzle with secondary fuel injectors located at the entrance to the afterburner.

From takeoff to full ramjet operation, the liquid oxygen - hydrogen rockets are operated stoichiometrically (O/F = 7.94) to prevent any burning in the mixer-diffuser section. The rocket exhaust and air are mixed and diffused to some higher temperature and pressure before entering the afterburner. Here additional fuel is injected to burn at an effective stoichiometric fuel to air ratio. The exhaust gases are then expanded through the nozzle.

In this study, the ratio of rocket propellant to ramjet captured air flow $\dot{w}_p/\dot{w}_a$ was held constant from takeoff to Mach 1 and decreased linearly with Mach number to 0 at the full ramjet operation Mach number M_r. The ratio $\dot{w}_p/\dot{w}_a$ was varied from 0.33 to

0.05 and M_r from 1.5 to 3.0 to maximize range. The full ramjet operation was shut down at SJ takeover at Mach 7.

The engines are located in the rear of the fuselage similar to the turboramjets and the same inlet operating characteristics as for the turboramjet were assumed. The engine weight was calculated from a constant factor of 125 pounds per square foot (610 kg/m^2) of capture area determined from reference 27.

Engine performance was determined from reference 28. For a capture area of 375 square feet (34.4 m^2), the installed engine thrust-to-weight ratio was 5.2.

Ejector dual-mode ramjet (EDMRJ). - This engine is basically an ejector ramjet and a SJ combined into one engine. As a result, this engine is used throughout the vehicle's speed range. From takeoff to full subsonic combustion ramjet operation M_r, the ejector dual-mode ramjet operates as an ejector ramjet. This being the case, parameters such as $\dot{w}_p/\dot{w}_a$ and M_r, discussed in the previous section, were also considered in sizing this engine to maximize the vehicle's range. From rocket shutdown M_r to Mach 7, the engine operates as a subsonic combustion ramjet. Above Mach 7, the engine operates as a supersonic combustion ramjet. Since the engine was located beneath the wing, the fuselage afterbody was changed from a cylinder to a cone to eliminate the large base drag in the transonic speed range.

The engine used in the study had an inlet contraction ratio (CR) of 10, the same as for the dual-mode ramjet of the TJ-DMRJ system. As pointed out for the DMRJ, with this CR only 10 percent of the full stream tube (captured mass flow ratio, 0.1) can be captured at Mach 1.0. Since insufficient thrust was available in the transonic speed range, some movable geometry was assumed. At Mach 1, the nominal capture mass flow ratio was increased by a factor of 2 ($m/m_c = 0.2$). This factor was reduced linearly to 0 at Mach 4 (same capture mass flow ratio at Mach 4 and above as for the fixed geometry DMRJ engine).

The weight of the installed engine was based on a value of 110 pounds per square foot (536 kg/m^2) of capture area as determined from reference 27. No specific weight penalty was included to account for the movable geometry. The decrement in range with increased engine weight was not investigated for this system since results similar to those obtained for the TJ-DMRJ (fig. 20) would be expected. The reason for this is that both SJ engines have the same inlet contraction ratio and, therefore, have similar performance. For an engine capture area of 75 square feet (6.97 m^2), the installed engine thrust-to-weight ratio is 7.3.

Engine Sizing

To maximize the range attained at each cruise Mach number, engines sizes and takeover or transition Mach numbers have been optimized for each propulsion system. This procedure is illustrated in appendix A for two of the more promising propulsion systems. The variation with Mach number of the thrust coefficient and the specific impulse for each of the five propulsion systems is given in appendix B.

RESULTS AND DISCUSSION

The range capability of each of the five propulsion systems considered in the study is presented first in terms of range to end of cruise and then to end of descent. Subsonic loiter times at end of descent are compared for two of the systems. The effect on range of sonic-boom overpressure, vehicle operating weight empty, SJ inlet and nozzle efficiency, chemical nonequilibrium expansion in the exhaust nozzle and cruise cooling requirements are determined.

Propulsion System 1

The variation in range to end of cruise for the TJ-R-SJ propulsion system is shown in figure 14 for cruise Mach numbers between 7 and 9. The maximum range attained was 2850 nautical miles (5.3×10^{6} m) for a cruise Mach number of 7.2. This range was achieved using six afterburning turbojets to accelerate the vehicle from takeoff to Mach 3.1. The turbojets are then shut down and two liquid oxygen - hydrogen rockets accelerate the vehicle from Mach 3.1 to 4. Four podded SJ's then power the vehicle during both the final acceleration and cruise phases of the flight. For a cruise Mach number of 7, the engine characteristics are as follows:

Turbojet:

Takeoff F/W_G, 0.5 (large enough for a satisfactory takeoff distance, although takeoff noise might demand some engine throttling and possibly engine resizing)

Rocket:

Total throat area, 1.6 square feet (0.149 m^2)
Design nozzle pressure ratio, 666
Equivalence ratio, φ, 1.0
Chamber pressure, 1000 psi (6.9×10^{6} N/m^2)
F/W_G at ignition, 1.0

Supersonic combustion ramjet:

Inlet contraction ratio, 6 ($\varphi = 0.3$ at Mach 4)
Capture area, 260 square feet (24.2 m^2)
F/W_G at Mach 4, 0.48
Engine performance, see appendix B

The poor range attained with this system can be explained by examining figure 15. This figure indicates the range attained and the fuel or propellant (as a percentage of the takeoff gross weight) expended during the acceleration and cruise phases of the Mach 8 flight. The vehicle's operating weight empty (OWE = takeoff weight minus propellant and payload) accounts for 50.6 percent of the takeoff gross weight (TOGW). Of this, 16.2 percent is propulsion system weight (TJ-R-SJ). Based on a 10-percent payload, this leaves 39.4 percent of the TOGW for propellant. Approximately one-half of this is consumed in the first 280 nautical miles (0.52×10^6 m) by the turbojets and the rockets while accelerating the vehicle from takeoff to Mach 4. However, the rocket is the biggest user consuming 57 percent of the propellant used between takeoff and Mach 4 while accelerating the vehicle from Mach 3.1 to Mach 4. Therefore, to minimize the effect of the rocket, the SJ is started at Mach 4. This requirement penalizes the SJ's performance since a low inlet contraction ratio (CR) is required to provide sufficient thrust while avoiding thermal choking (see fig. 8). This problem could be relieved by increasing the combustor area ratio. However, this was not investigated because of the poor range attained by this system relative to several others. By the time the vehicle reaches Mach 8, 83.5 percent of the propellant load has been used. Based on the ground rule that 10 percent of the hydrogen fuel load is for reserve, only 3.6 percent of the TOGW is available as fuel for cruise. The lift-drag ratio and the specific impulse at cruise are 5.6 and 2058 seconds. Thus, the principal reasons for propulsion system 1 achieving a maximum range of only 2850 nautical miles (5.3×10^6 m) are (a) the weight of the propulsion system, (b) the low performance of the rocket, and (c) the low performance of the SJ in comparison to a higher CR engine.

The poor range attained with propulsion system 1 could probably be improved somewhat by increasing the combustion chamber area ratio of the SJ. However, to relieve the problems associated with points (b) and (c), the rockets of system 1 were replaced with subsonic combustion ramjets of propulsion system 2.

Propulsion System 2

System 2 consists of six turboramjets and four podded SJ's. The range attained to end of cruise with these engines is shown in figure 16 for cruise Mach numbers between

8 and 12. The maximum range attained to end of cruise was 5995 for a cruise Mach number of 8. The engine characteristics which resulted in this range are as follows:

Turboramjet:

(1) See ANALYSIS section (p. 5) for detail characteristics.

(2) Thrust-to-gross-weight ratio at takeoff, 0.471; this value resulted in a 16-percent transonic thrust margin (F - D)/D (see appendix A for sizing of engines).

Supersonic combustion ramjet:

(1) Capture area, 160 square feet (14.9 m^2); for Mach 12, the engine capture area had to be increased to 200 square feet (18.6 m^2) because of the decrease in SJ thrust coefficient with increasing Mach number.

(2) Inlet contraction ratio, 25; this value was used for a SJ takeover Mach number of 6.5. (See appendix A for the method used in sizing engines and appendix B for engine performance.)

At higher cruise speeds, the range decreases; in fact, the rate at which it declines increases with cruise speed. For example, the range decreases by 225 nautical miles (0.417×10^{3} m) when the vehicle cruises at Mach 10 rather than 8. If the cruise speed is increased from Mach 10 to 12, the decrement in range is 890 nautical miles (1.65×10^{6} m). The next two figures indicate the reason for this sharp decrease.

Figure 17 indicates the variation in (1) the fuel available for cruise, (2) the overall engine efficiency η_e, and (3) the aerodynamic efficiency L/D of the vehicle for cruise Mach numbers between 8 and 12. Although figures 17(b) and (c) indicate the possibility of a 14-percent increase in range by cruising at Mach 12 rather than 8 (as calculated by the Breguet range equation), figure 17(a) indicates that there is very little fuel available for cruise by the time the vehicle accelerates to Mach 12. Thus, the Mach 12 vehicle is not really a cruise, but a boost-descent vehicle. Most of its 4880 nautical miles (9.04×10^{6} m) range to end of cruise is achieved during the acceleration portion of the flight. However, this is not true of the Mach 8 vehicle.

For the Mach 8 cruise vehicle, figure 18 indicates the range achieved and the fuel expended during the climb and cruise portions of the flight. In contrast to the Mach 12 vehicle which has about 1 percent of the TOGW avaiable as fuel for cruise, the Mach 8 vehicle has over 13 percent. Thus, its cruise range (4076 n mi; 7.56×10^{6} m) is almost equal to the total range attained by the Mach 12 vehicle (4880 n mi; 9.04×10^{6} m).

In comparison to the TJ-R-SJ vehicle, the range capability of the present vehicle is vastly superior. For a cruise Mach number of 8, both vehicles have about equal amounts of fuel or propellant onboard at takeoff. With the TJ-R-SJ system, the propellant accounts for 39.4 percent of the TOGW as compared to 37.5 percent for the TRJ-SJ system. The present vehicle travels 1919 nautical miles (3.55×10^{6} m) while accelerating to

Mach 8 as compared to 1764 nautical miles (3.27×10^6 m) for the TJ-R-SJ system. However, because of the greater overall specific impulse of the present system, only 54.5 percent (20.4 percent of TOGW) of the fuel load is consumed in reaching Mach 8 as compared to 83.5 percent (32.8 percent of TOGW) with the TJ-R-SJ. Based on the fuel reserve requirements (10 percent of the hydrogen fuel), the present system has 3.6 times as much fuel available for cruise. In addition, the cruise specific impulse of the SJ is about 30 percent greater (2700 sec) because of the higher inlet contraction ratio (25 against 6). For these reasons, the vehicle using the present propulsion system attains a range to end of cruise of 5995 nautical miles (11.10×10^6 m) for a cruise Mach number of 8 as compared to 2730 nautical miles (5.06×10^6 m) with the TJ-R-SJ.

Propulsion System 3

Propulsion system 3 consists of six turbojets and four podded dual-mode ramjets. As indicated in the Engine Cycles section (p. 9) and appendix B, significant improvements in the system's thrust coefficient can be obtained with variable inlets on the ramjets and transonic thrust augmentation by the ramjets. To assess these effects on the vehicle's range capabilities, the following variations in ramjet operation were considered:

Scheme A: No ramjet augmentation - fixed geometry. Ramjet operation started at turbojet shutdown at Mach 3.1.

Scheme B: Ramjet augmentation - fixed geometry. The ramjet operation started at Mach 1.2 to augment the thrust of the turbojets. Bypass doors in the inlet were operated up to Mach 4 to bypass excess air to prevent unstarting.

Scheme C: Ramjet augmentation - variable inlet. Ramjet operation started at Mach 1.2. The inlet throat was assumed to vary up to Mach 4 to prevent unstarting after which the inlet contraction ratio was fixed at 10.

Since there is little information with which to determine the weights of movable structures at hypersonic speeds, no weight increases due to the movable structure of scheme C were included in the calculations. Therefore, scheme C may be viewed as an upper limit of the range capabilities of scheme B. However, the decrease in range caused by assuming various weight penalties was studied and will be discussed later.

The range to end of cruise for Mach numbers ranging from Mach 8 to 12 is shown in figure 19 for the three types of ramjet operation. The range is seen to drop rapidly after a cruise Mach number of 10 because the ramjet engine size and weight increases with cruise Mach number decreasing the fuel available for cruise. For example, the ramjet engine size required for Mach 12 cruise is 260 square feet (24.15 m^2), whereas for

Mach 10 it is 180 square feet (16.72 m^2). Thus, the drop in range by cruising at Mach 12 instead of 10 is 850 nautical miles (1.57×10^6 m) for scheme A, 550 nautical miles (1.02×10^6 m) for scheme B, and 660 nautical miles (1.22×10^6 m) for scheme C. The maximum ranges were attained at a cruise Mach number of 8 and are as follows: A - 5225 nautical miles (9.68×10^6 m), B - 5860 nautical miles (10.85×10^6 m), and C - 6650 nautical miles (12.32×10^6 m).

The advantages of ramjet augmentation are readily seen by the difference in the range capabilities of schemes A and B, where an increase of 635 nautical miles (1.18×10^6 m) results from augmentation. With the optimistic assumption of no weight increase due to a variable inlet, the increase in range due to the higher thrust coefficients of a variable inlet is the difference between the ranges of B and C, which is about 800 nautical miles (1.48×10^6 m) at a cruise Mach number of 8. However, this advantage is decreased if the engines are heavier due to the variable structure. This effect is shown in figure 20. For this figure the increase in range over that of scheme B is shown for various assumptions of increased engine weight due to movable inlet structures. As mentioned before, the increase in range with no weight penalty is about 800 nautical miles (1.48×10^6 m) for a cruise Mach number of 8. According to reference 25, the engine weight could increase by 50 percent or more due to a variable geometry inlet. With a 50-percent penalty the advantages in a movable inlet are considerably reduced. For example, at Mach 8 cruise the increase in range is now only 340 nautical miles (0.63×10^6 m) above a fixed inlet. The lower limit of the shaded area in figure 19 represents the range capabilities using transonic ramjet augmentation and a 50-percent increase in engine weight due to a variable inlet. It is seen that this curve is approaching that of scheme B. At Mach 12 it is seen that with this engine weight assumption the range obtained with a variable inlet ramjet is less than that of the fixed inlet engine. In view of these results, the fixed inlet engine using transonic augmentation (scheme B) is the most desirable scheme for propulsion system 3 and the remaining discussion will be restricted to this scheme.

The engine characteristics that maximized range at Mach 8 for this system are as follows:

Turbojet:

- See ANALYSIS section (p. 13) for detailed characteristics
- Thrust-to-gross-weight ratio at takeoff, 0.42
- Transonic thrust margin with augmentation, 11 percent
- See appendix A for engine sizing

Supersonic combustion ramjet:

- Capture area, 200 square feet (18.58 m^2)

Figure 21 shows range against fuel consumption in percent of gross takeoff weight. The acceleration range is 1260 nautical miles (2.33×10^6 m) which is 700 nautical miles (1.30×10^6 m) less than system 2, but the fuel fraction is 19.6 percent, which is about equal to that of system 2 (20.4 percent). The reason for this may be seen by comparing the thrust coefficients and specific impulses of systems 2 and 3 in appendix B (figs. 39 and 40(a)). Between Mach 1.2 and 3.1 system 3 has higher thrust coefficients than 2 but lower specific impulses because of ramjet augmentation. As a result system 3 accelerates faster, therefore covering less range while using about the same amount of fuel (39 percent of the acceleration fuel). Between Mach 3.1 and 8 where 86 percent of the acceleration range is covered and 61 percent of the acceleration fuel is used, the thrust coefficients and specific impulses of system 3 are lower than those of system 2. Between Mach 3.1 and 6.5 the values for system 2 are higher because of the variable geometry of the TRJ. Above Mach 6.5 this is due to the higher inlet contraction ratio of the SJ.

For example, the system 2 thrust coefficients range from 20 percent higher than those of system 3 at Mach 5 to 5 percent higher at Mach 8. However, system 3 still has a higher rate of acceleration because of the larger SJ engine and, therefore, a greater net thrust.

When the propulsion weight fraction and total fuel weight fractions of systems 2 and 3 in figures 18 and 21 are compared, it is seen that the system 3 propulsion weight is 13.2 percent of the gross takeoff weight which is 3 percent less than the 16.2 percent of system 2. The total fuel fraction is 3.7 percent higher than that of system 2. The vehicle weight breakdown is discussed in a later figure (fig. 26). Thus system 3 has more fuel available for cruise which tends to offset the lower specific impulse and results in the range capabilities of systems 2 and 3 being about the same.

Propulsion System 4

This system consists of six ejector ramjets which power the vehicle from takeoff to Mach 7 after which four podded SJ's are used.

Figure 22 shows range to end of cruise for this system. As in the previously discussed systems, the range decreases rapidly after a cruise Mach number of 10 due to the large engines needed to accelerate to Mach numbers above 10. The range maximized at Mach 8 at 4310 nautical miles (7.98×10^6 m). The engine characteristics for this condition are the following:

Ejector ramjet:

Rocket propellant/ramjet airflow, 0.05

Pure ramjet operation, begins at Mach 1.5

Capture area, 380 square feet (35.3 m^2)

Takeoff thrust-to-gross-weight ratio, 0.48

Transonic thrust margin, 18 percent

Supersonic combustion ramjet:

Capture area, 180 square feet (16.7 m^2)

Compared to the TRJ-SJ and TJ-DMRJ systems, the range capabilities of this system are quite low even though the propulsion system accounts for only 12.3 percent of the TOGW as compared to 16.2 and 13.2 for the TRJ-SJ and TJ-DMRJ systems. The reason for this is seen in figure 23 which presents the range against expended fuel or propellant fraction. From takeoff to Mach 1.5 during ejector ramjet operation, a propellant weight fraction of 19.6 percent is expended or 46 percent of the total propellant weight. However, only 50 nautical miles (92.6×10^3 m) in range is covered or about 1.15 percent of the total range. This reflects the low specific impulse and the greater thrust margin of the ejector ramjet compared to the turbojet and turboramjet.

After Mach 1.5 during pure ramjet operation, the range is seen to increase rapidly with fuel fraction; however, only 800 nautical miles (1.48×10^6 m) acceleration range is covered and a fuel plus liquid oxygen weight fraction of 29.6 percent is consumed. Thus, about 70 percent of the total propellant weight is consumed during acceleration; this is high compared to the TRJ-SJ and TJ-DMRJ systems which used 55 to 60 percent for acceleration. Thus, system 4 has only a 9.7-percent propellant weight fraction left for cruise compared to systems 2 and 3 which have of the order of 13 to 17 percent fuel fraction for cruise. Therefore, system 4 not only attains less acceleration range for a greater expenditure of propellant weight but has less fuel for cruise than systems 2 and 3. The unattractive range capabilities of this system are therefore the result of the low specific impulse characteristics of the ramjet during ejector ramjet operation.

Propulsion System 5

The last propulsion system, the ejector dual-mode ramjet, is the only single engine system of the five systems considered. The range attained to end of cruise with this engine is shown in figure 24 for cruise Mach numbers between 8 and 10. The maximum range was attained for a cruise Mach number of 8. In this case, the range to end of cruise was 5320 nautical miles (9.85×10^6 m). Increasing the cruise Mach number from 8 to 10 results in an 8-percent decrease in range. For Mach 8, the total engine capture area for the four engines was 300 square feet (27.9 m^2). The corresponding sea level static thrust to gross weight ratio was 0.481.

In the current study, the maximum range was attained by operating the engines at a primary to secondary flow rate ratio of 0.05 from takeoff to Mach 1. From Mach 1 to 2, this ratio was decreased linearly to 0; therefore, the engines operated as subsonic combustion ramjets at Mach 2. For a given size engine, a higher flow rate ratio, such as 0.33, means more propellant is being used by the rocket. This increases the thrust margin but decreases the overall specific impulse during the augmented rocket mode of operation with the result that less fuel is available for cruise.

If the augmented mode of operation is terminated at Mach 1.5, the thrust margin in the transonic speed range is reduced and a larger engine is required. Thus, the range attained decreases although the overall specific impulse is increased. The range is also penalized if the augmented mode of operation is extended to Mach 3. For this case, more propellant is used because of the rocket's low specific impulse. From Mach 2 to 7, the ejector dual-mode ramjet operates as a subsonic combustion ramjet. Above Mach 7, the combustion mode changes from subsonic to supersonic.

The propellant expended and the range attained during the acceleration and cruise portions of the Mach 8 flight are indicated in figure 25. Since the present propulsion system does not include any heavy turbomachinery, its weight accounted for only 6.6 percent of the TOGW. As a result, the OWE of this vehicle accounted for only 43.9 percent of the TOGW as compared to 52.5 percent for the vehicle using the TRJ-SJ propulsion system. Thus, at takeoff 46.1 percent of the TOGW is propellant. Almost 38 percent of this (17.4 percent TOGW) is used during the augmented rocket mode of operation, that is, from takeoff to Mach 2. During this period, the vehicle travels 100 nautical miles (185.2×10^3 m). At the start of cruise at Mach 8, the vehicle has traveled 600 nautical miles (1.11×10^6 m) and expended 56.5 percent of the onboard propellant (26.1 percent of TOGW). Based on the reserve requirements, 16 percent of the TOGW is available as fuel for cruise as compared to 14 percent with the TRJ-SJ system.

However, the advantage of this extra fuel is partially offset by the lower cruise specific impulse, 2370 against 2700 for the TRJ-SJ system. This results from using an inlet contraction ratio of 10 for this system as compared to 25 for the SJ of the TRJ-SJ system. With a contraction ratio of 10, the engines operate at an equivalence ratio during cruise of 0.472 which satisfies the engine cooling requirements. With a contraction ratio of 25, an equivalence ratio in excess of that required for cruise thrust would be required to cool the engines. This would lower the effective specific impulse of the higher contraction ratio engine. In comparison to the other four propulsion systems, the maximum range achieved to end of cruise with the EDMRJ is about 700 nautical miles (1.30×10^6 m) less than attained with the TRJ-SJ and TJ-DMRJ and 1010 and 2470 nautical miles (1.87 and 4.57×10^6 m) greater than the ERJ-SJ and TJ-R-SJ. However, these numbers do not include any weight penalty to account for the variable geometry feature of the engine. Assessing such a penality would only detract from the system.

Weight Breakdown

To this point each of the systems has been compared on the basis of range to end of cruise. Figure 26 compares each of the five propulsion systems in terms of its vehicle weight breakdown for a cruise Mach number of 8. Since the payload was equal to 10 percent of the TOGW for each of the five systems, the remaining weight is composed of the propulsion system, the OWE minus the propulsion system, liquid oxygen (if required), and hydrogen fuel.

The weight of the propulsion system varies from a low of 6.6 percent of the TOGW for the EDMRJ to a high of 16.2 percent for the TJ-R-SJ and the TRJ-SJ.

The OWE minus the propulsion system weight varies from a low of 34.4 percent for the TJ-R-SJ vehicle to a high of 37.3 percent for the EDMRJ. This weight depends on items such as fuel storage, volume required for the initial acceleration engines, insulation requirements, and fuselage geometry. Since vehicles 1 and 4 require less storage volume and insulation, the OWE minus the propulsion system weight fraction for these vehicles is lower. With regard to vehicles 3 and 4, vehicle 4 has a smaller propellant volume since liquid oxygen has a higher density than liquid hydrogen. Also, vehicle 4 spends less time at cruise; therefore, it has less insulation. Hence a lower OWE - propulsion system weight fraction would be expected for system 4. For vehicle 5, this weight fraction is influenced by the fact that the fuselage afterbody is conical rather than cylindrical. Decreasing any one of these weights (payload, propulsion system or OWE minus propulsion system weight) would result in an increase in the fuel available for cruise and, therefore, a greater range. After expending the fuel available for cruise, the vehicle is ready to descend.

Descent

At hypersonic speeds the vehicle possesses a great deal of energy, kinetic plus potential. Therefore, the descent portion of the mission can represent a significant portion of the total range. For this study, a gliding descent was considered. During descent the podded engines were blocked off by plates which were hinged at their leading edge (angle with wing equals 20°). The lift and drag associated with these plates were included in the vehicle's performance. A representative flight path was picked for the descent (fig. 27(a)). This path followed a fairly flat trajectory so as to try and maximize the descent range while not exceeding the metal temperature limits established at the cruise conditions. An indication of the range attained with this representative path as compared to the maximum attainable is indicated by the comparison of the operating L/D ratio and the maximum L/D ratio (fig. 27(b)).

In figure 28, the descent range has been added to the maximum range attained to end of cruise with each of the five propulsion systems. In all cases, the addition of the descent range shifts the optimum cruise Mach number to a higher value. With the addition of the descent range, the vehicles range at end of cruise is increased by roughly 1400 nautical miles (2.59×10^6 m) at Mach 10 and 750 nautical miles (1.39×10^6 m) at Mach 8. In addition, the optimum cruise Mach number is increased from 8 to about 10.5 for the TRJ-SJ and 11.5 for the TJ-DMRJ while the range for both systems is increased from about 6000 to about 7150 nautical miles (11.11 to 13.24×10^6 m).

The descent portion of the flight affects the vehicle's block speed. Block speed is defined as the total range divided by the total time. Since the direct operating cost of a commercial transport is inversely proportional to block speed, a more detailed study of the mode of descent (gliding against propulsive) should be made.

Loiter

In addition to a required range capability, a hypersonic transport must also be able to loiter at subsonic speeds. As shown previously, the range obtained with the EDMRJ was about 6000 nautical miles (11.1×10^6 m). Therefore, on the basis of range, this engine would have to be considered as a contender for propelling a hypersonic transport. However, figure 29 indicates the disadvantage of the lower performance although lighter weight of a hybrid rocket propulsion system (such as the EDMRJ) compared to the higher performance of the heavier turbomachinery. When the data for this figure were calculated, it was assumed that all of the reserve propellant except for 5000 pounds (2268 kg) was used for loiter. During loiter the vehicle cruised at an altitude of 36 000 feet (10 973 m) and at a Mach number of 0.9. The loiter capability of the vehicle using the turbomachinery, in this case the TRJ (system 2) operating as a nonafterburning turbojet, is an order of magnitude greater than that attained with the hybrid ejector dual-mode ramjet engine (system 5) operating as an augmented rocket. Thus on the basis of loiter capability, the EDMRJ and therefore the ERJ-SJ propulsion systems do not appear attractive for transport application. Therefore, of the five propulsion systems initially considered, the TRJ-SJ and the TJ-DMRJ appear to be the most promising for a hypersonic cruise transport. Of these two promising systems, the vehicle using the TRJ-SJ system was subjected to a sensitivity study.

Range Sensitivity

A sensitivity study was performed for the Mach 10, TRJ-SJ cruise vehicle to determine the effects of sonic-boom overpressure, vehicle operating weight empty (OWE), supersonic combustion ramjet inlet and nozzle efficiency, chemical nonequilibrium expansion (kinetic effect) in the exhaust nozzle, and cruise cooling requirements on range.

Sonic boom. - As indicated in figure 30(a), one of the problem areas created by both supersonic and hypersonic flight is sonic-boom overpressure. Unfortunately, this problem is most severe in the transonic speed range or during the transonic thrust-drag pinch. In the present study, the ground overpressure was limited to 2.5 pounds per square foot (119.7 N/m^2). This limit altered the representative flight path between Mach 1 and 2.75. From Mach 1 to Mach 2.75, the vehicle traveled about 150 nautical miles (277.8×10^3 m) (fig. 30(b)). Above Mach 2.75, the sonic-boom overpressure decreases with increasing Mach number. In fact, the overpressure decreases to 0.5 pound per square foot (23.9 N/m^2) at Mach 10 at the start of cruise. Thus, for cruise, the sonic-boom overpressures were below the current goal of 1.7 pounds per square foot (81.4 N/m^2) for the SST; therefore, no definite value for cruise was considered in the present study. Unfortunately, while accelerating to Mach 10, the vehicle traveled about 4500 nautical miles (8.3×10^6 m). However, the vehicle traveled only about 350 nautical miles (0.65×10^6 m) before sonic-boom levels below the initial cruise sonic-boom level of the long range SST (1.7 lb/ft^2) (81.4 N/m^2) were achieved. The transonic ground sonic-boom overpressure may be reduced by increasing the altitude; however, a higher flight path decreases the thrust margin. This, then increases the climb fuel requirements and/or the engine size. Thus, for a fixed TOGW airplane, any reduction in transonic sonic-boom overpressure will be accompained by a decrease in total range.

Figure 31 indicates the effect of the climb transonic sonic-boom overpressure limit on range to end of descent. If the overpressure limit is increased by 5 percent (2.62 lb/ft^2) (125.45 N/m^2) the range is increased by 1.2 percent. Decreasing the limit by 5 percent (2.38 lb/ft^2) (113.7 N/m^2) results in a 1.0-percent decrease in range. However, below a limit of 2.3 the range drops drastically, and it is seen that limits below 2.2 pounds per square foot (105.3 N/m^2) are unattainable. This is due to the rapid increase in engine size and weight necessary below limits of 2.3. A more moderate decrease in range with decreasing transonic overpressure may result with a nonturboaccelerator type engine such as the ERJ. This is because engines of this type tend to have higher thrust to weight ratios than turboaccelerators and their thrust is less sensitive to changes in altitude.

Supersonic-combustion-ramjet cooling requirements. - Earlier it was shown (see Supersonic combustion ramjet (SJ), p. 10) that the equivalence ratio at which the SJ's operated at cruise was adequate for cooling the engines. However, if the wall temperature must be maintained at less than 2000° R (1110° K) or if the heat-transfer efficiency is less than 85 percent, then a higher equivalence ratio would be required. Figure 32(a) indicates that increasing the cruise equivalence ratio from that required to achieve the maximum Breguet factor (0.76) to 0.95, an increase of 25 percent, results in a decrement in range of only 1.7 percent. This decrease is small because only about 11 percent of the total range to end of descent is due to the cruise portion of the flight. For cruise Mach numbers below 10, the cruise cooling requirements have a greater effect since the cruise range accounts for a larger portion of the total range. For Mach 8, the range to end of descent was decreased by 5.5 percent as a result of operating the engines at an equivalence ratio of 0.95 rather than 0.71 which resulted in the maximum Breguet factor.

Operating weight empty (OWE). - A parameter which has a greater effect on the range is the vehicle's operating weight empty. Figure 32(b) indicates the effect of the OWE on the range to end of descent. If the OWE had been 0.494 rather than the calculated value of 0.519, a reduction of about 5 percent, the range to end of descent would have been increased by 11 percent. Thus, new structural techniques now being studied (ref. 29) can have a significant effect on the range of a hypersonic transport.

Inlet, nozzle, and kinetic effects. - The last three studies determined the effect of SJ inlet pressure recovery (see fig. 33), nozzle efficiency (fig. 34), and nozzle kinetic effects (fig. 35) on range for cruise Mach numbers between 8 and 10. For each Mach number, a fixed airplane was used. The vehicle was propelled by the TRJ-SJ propulsion system.

To determine the effect of SJ pressure recovery on range, the reference pressure recovery schedule (curve A) shown at the top of figure 33(a) was reduced by 25 percent resulting in schedule B. As indicated previously the pressure recoveries of schedule A represent a goal for a SJ inlet. The effect of this new pressure recovery schedule on the airplane's range is shown in figure 33(b). For a cruise Mach number of 8, the range of the reference vehicle was decreased by 2 percent when the SJ inlets achieved pressure recoveries indicated by schedule B rather than the design values of schedule A. When the Mach 10 reference vehicle operated with schedule B, the vehicle drag exceeded the engine thrust. To overcome this, the SJ capture area was increased from the optimum 160 to 180 square feet (14.9 to 16.7 m^2). Thus, the range shown at Mach 10 for schedule A is lower than the range (dashed curve) attained with the optimum sized engine. The range attained with schedule B was 9.6 percent lower than with schedule A using the optimum sized engine. The effect on a fixed airplane of missing the design pressure recovery by 25 percent can be drastic.

To determine the effect of the exhaust nozzle efficiency on range, the nozzle efficiency (K_N) was decreased from the nominal value of 0.928 to a value of 0.89. This 4-percent decrease resulted in losses of 4.4 and 5.8 percent in range to end of descent for cruise Mach numbers of 8 and 10, respectively (fig. 34). Thus, a 4-percent decrease in the exhaust nozzle efficiency (K_N) results in about a 5-percent decrease in range.

Another item related to the exhaust nozzle which will influence the performance of the airplane is the exhaust nozzle kinetic or dissociation effects. As indicated in figure 35(a) the kinetic effects increase with Mach number. This loss in engine performance is due to the fact that part of the heat of reaction is not released during the expansion process. This effect increases with Mach number because of the decrease in the nozzle pressure and the associated increase in the recombination time or period. At Mach 8, the specific impulse is reduced by 30 seconds as compared to 190 seconds at Mach 10. This decrease in specific impulse will result in 2- and 5.5-percent decreases in range to end of descent for cruise Mach numbers of 8 and 10 (fig. 35(b)).

CONCLUDING REMARKS

Because the supersonic combustion ramjet (SJ) does not possess a static thrust capability, an additional engine (or engines) is required to initially accelerate a SJ powered vehicle. A mission study was, therefore, made to determine which of several engines would be best for accelerating a hydrogen-fueled SJ cruise transport from takeoff to SJ takeover. The comparison of each engine (or engines) in combination with the SJ was based on range. Since the results are highly dependent on the assumptions used to define the various subsystems, the reported range capabilities and the relative performance of the studied propulsion systems should be recognized as only preliminary estimates.

Of the five engine systems considered, the two most promising were (a) the turboramjet - SJ and (b) the turbojet - dual-mode ramjet. The SJ of system (a) operates from Mach 6.5 to end of cruise. The dual-mode ramjet of system (b) operates as a subsonic combustion ramjet from Mach 1, where it augments the turbojet, to Mach 7. Above Mach 7, this engine operates in the supersonic combustion mode. Both of these engine systems attained a maximum range to end of descent of about 7100 nautical miles (13.1×10^6 m) for a cruise Mach number of approximately 10.

The descent range was based on a gliding descent. However, a propulsive descent may be required to either cool the engines or remove residual heat. These cooling requirements will determine the thrust level to which the engine may be throttled during descent.

For the TRJ-SJ system, the performance of the TRJ was based on an inline engine. Another version of this engine which deserves consideration is the wraparound TRJ.

For the TJ-DMRJ system, the maximum range was achieved with a dual-mode ramjet having an inlet contraction ratio of 10. With this relatively low inlet contraction ratio, the supersonic combustion performance of the engine is penalized. Thus, any idea, such as thermal precompression or variable geometry, which may possibly improve this performance should be studied.

In sizing the initial acceleration engines, a transonic sonic-boom limit of 2.5 pounds per square foot (119.7 N/m^2) was used. However, engine noise during and immediately after takeoff was not considered in the present study. Since this may affect final engine sizing, fuel requirements, and range, it should be considered in a more detailed study.

At hypersonic speeds, the sonic-boom level may be lower than that of the SST. This is mainly due to the higher altitude associated with hypersonic vehicles. At Mach 10, for example, the sonic-boom overpressure was computed to be about 0.5 as compared to 1.7 pounds per square foot (23.4 as compared to 81.4 N/m^2) for the SST cruising at Mach 2.7. This could be a significant advantage for the hypersonic transport. However, to exploit this advantage the acceleration range while at supersonic speeds must be shortened. During this phase of the flight the sonic-boom overpressure is comparable to that of the SST. Therefore, oversized engines for the initial acceleration phase of the flight and/or two-stage configurations should be considered.

The sensitivity of range to various parameters was determined for the delta-wing body configuration using the TRJ-SJ propulsion system to a cruise Mach number of 10. The parameters considered were sonic-boom overpressure, vehicle operating weight empty, SJ inlet pressure recovery, SJ exhaust nozzle efficiency, and nozzle kinetic effects.

With the nominal values for these parameters, the range to end of descent was 7140 nautical miles (13.2×10^6 m). A 5-percent decrease in the nominal value (2.5) of the transonic sonic-boom overpressure results in a 1-percent decrease in range. A 5-percent decrease in the OWE means an 11-percent increase in range. Since the other parameters affected the performance of the SJ over its entire operating range, a fixed airplane (same weight breakdown as for nominal case) was used in determining their effect on range. A 25-percent reduction in the nominal pressure recovery schedule resulted in a 9.6-percent decrease in range. Decreasing the nominal exhaust nozzle efficiency by 4 percent resulted in a 5.8-percent decrease in range. Accounting for kinetic effects in the exhause nozzle decreases the range by 5.5 percent. Thus, it appears that for the parameters investigated the largest gains in range can be achieved by (1) developing new structural techniques which result in lighter weight vehicles and engines, and (2) attaining maximum nozzle performance.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 19, 1967,
126-15-03-08-22.

APPENDIX A

ENGINE SIZING

To maximize the range attained with any propulsion system, each engine has to be correctly sized. The procedure used in sizing the engines for two of the more promising propulsion systems considered in this study is outlined.

Turboramjet-SJ

An iterative procedure involving four independent variables was required to determine final engine sizes at each cruise Mach number for this system. The four variables were (1) takeoff thrust-to-weight ratio, (2) SJ capture area, (3) SJ takeover Mach number, and (4) SJ inlet contraction ratio. The results obtained for this system and a cruise Mach number of 8 are shown in figure 36. Initially, the takeoff thrust-to-weight ratio was varied for fixed values of the other three variables (fig. 36(a)). After sizing the TRJ to maximize the range for these conditions, the procedure was repeated for other values of SJ capture area. A resultant curve is shown in figure 36(b). The entire procedure was then repeated for SJ takeover Mach numbers of 5.0 and 7.0. As indicated in figure 36(b), the range is greater for a SJ takeover Mach number of 6.5, a SJ capture area of 160 square feet and an inlet contraction ratio of 25. With the other three variables fixed, the inlet contraction ratio was varied between 10 and 25.

After the values of the four independent variables giving the greatest range for a cruise Mach number of 8 were found, the iterative sizing procedure was repeated for other values of cruise Mach number.

Turbojet - Dual-Mode Ramjet

An engine sizing routine similar to that used for the TRJ-SJ was also used for the TJ-DMRJ for each cruise Mach number to maximize the range. Four independent variables were again involved in arriving at the final engine sizes. The four variables were (1) the takeoff thrust-to-weight ratio, (2) the DMRJ capture area, (3) the DMRJ conversion Mach number (conversion from subsonic to supersonic combustion), and (4) the DMRJ inlet contraction ratio. The results obtained for this system and a cruise Mach number of 8 are shown in figure 37.

The takeoff thrust-to-weight ratio was varied to maximize the range for fixed values of the other three variables (fig. 37(a)). After sizing the TJ for these conditions, the procedure was repeated for other values of DMRJ capture area. The preceding routine results in a curve such as the one shown in figure 37(b). For conversion Mach numbers of 5 and 6, new curves are generated. The maximum range was obtained for a conversion Mach number of 7, a DMRJ capture area of 210 square feet (19.5 m^2) and an inlet contraction ratio of 10. With the other three variables fixed, inlet contraction ratio was varied between 8 and 15. After the maximum range was achieved for a cruise Mach number of 8 by varying in an orderly manner the four independent variables, the iterative sizing routine was repeated for other cruise Mach numbers.

APPENDIX B

PROPULSION SYSTEM PERFORMANCE

Typical values of the net thrust coefficient and specific impulse for each of the five propulsion systems considered in the study are indicated in figures 38 to 42. The performance includes the effect of operating the engines over the nominal flight path with the inlets in the wing pressure field. For the airbreathing engines, the thrust coefficients are based on the total operating engine capture area (downstream of wing shock) and the free stream dynamic pressure. The performance includes the additive drag associated with the off-design performance of the inlet, that is, operating at a capture mass flow ratio of less than 1.0. The thrust coefficient of the rocket is based on a combustor chamber pressure of 1000 psi (6894.76×10^3 N/m^2) and a throat area of 1.6 square feet (0.15 m^2).

Propulsion System 1 (TJ-R-SJ)

Propulsion system 1, (fig. 38) used three different engine cycles to accelerate the vehicle to the desired cruise Mach number. Six turbojets accelerated the vehicle from takeoff to Mach 3.1. At Mach 3.1, the turbojets were shut down and rockets accelerated the vehicle from Mach 3.1 to 4. From Mach 4 to cruise, four podded SJ's powered the vehicle.

The performance for the turbojet and the rocket is based on the engine characteristics listed in the ANALYSIS (pp. 13 and 14) section.

To avoid thermal choking in the present SJ and obtain sufficient thrust for accelerating the vehicle, an inlet contraction ratio (CR) of 6 was used. For this contraction ratio, the equivalence ratio was varied from 0.3 at Mach 4 to 1 at Mach 8. Other characteristics for the SJ are listed in the ANALYSIS (p. 10) section. For a cruise Mach number of 9, the thrust coefficient and specific impulse at cruise were 0.4 and 1845 seconds.

Propulsion System 2 (TRJ-SJ)

The turboramjet of propulsion system 2 operated as an afterburning turbojet from takeoff to Mach 3.1. Above 3.1, the air is bypassed around the turbomachinery to the afterburner. The engine then operates as a subsonic combustion ramjet from Mach 3.1 to SJ takeover, Mach 6.5. The performance for the Mach 10 cruise vehicle is given in figure 39.

The performance shown for the SJ in figure 39 is for an inlet contraction ratio of 25. For a cruise Mach number of 10, the thrust coefficient and specific impulse at cruise were 0.76 and 2270 seconds.

Propulsion System 3

With this propulsion system, six turbojets accelerate the vehicle from takeoff to Mach 3.1. Above Mach 3.1 the turbojets are shut down and four fixed geometry dual-mode ramjets operate as the main propulsion system. This engine operates as a subsonic combustion ramjet from Mach 3.1 to 7. Beyond Mach 7, the engines operate in the supersonic combustion mode. In the present study, augmentation of the turbojets with the dual-mode ramjets in the transonic speed range (Mach 1.2 to 3.1) was considered.

The effect on the thrust coefficient and the specific impulse of augmenting the turbojets with a fixed geometry (CR = 10) dual-mode ramjet is shown in figure 40(a). The thrust coefficient is based on the capture area of the turbojet. It is seen that the thrust coefficient is increased from 3 to 4 at Mach 1.2 and almost doubled at Mach 3; however, the specific impulse is decreased. As indicated in the ANALYSIS section (p. 16), for an inlet contraction ratio of 10 the fixed geometry inlet captures only 10 percent of the available mass flow transonically. This percentage could be increased appreciably by incorporating variable geometry in the inlet. In the present study, variable geometry was used to increase the capture mass flow ratio from 10 to 60 percent transonically (fig. 13).

The effect on the thrust coefficient and the specific impulse of augmenting the turbojets with a movable geometry ramjet is shown in figure 40(b). The overall specific impulse is seen to decrease transonically but increase between Mach 2 and 3. On the other hand, the thrust coefficient is increased by a factor of about 2.5 between Mach 1.2 and 3. For a cruise Mach number of 10, the thrust coefficient and the specific impulse at cruise were 0.75 and 1750 seconds. Augmentation is one method of possibly decreasing the weight of the overall propulsion system. Another method is to replace the turbojet with an engine like the ejector ramjet as in the next propulsion system.

Propulsion System 4 (ERJ-SJ)

Propulsion system 4 consists of ejector ramjet and SJ engines. The ejector ramjets operate as augmented liquid oxygen - hydrogen rockets from takeoff to Mach 1.5. From takeoff to Mach 1, the ratio of the rocket propellant weight flow to the inlet air flow ($\dot{w}_P/\dot{w}_a$) is held constant at a value of 0.05. Between Mach 1 and 1.5 this value is decreased linearly to 0. The schedule used for the captured airflow was the same as for the TRJ. The inlet captures a full stream tube starting at Mach 4.5. At Mach 1, 30 percent of the full stream tube is captured. The ejector ramjet operates as a subsonic combustion ramjet between Mach 1.5 and 7. The ramjet is operated at a stoichiometric fuel-air ratio. Figure 41 indicates the variation in the specific impulse and the thrust coefficients. This performance was based on the data of reference 28.

Beyond Mach 7, the vehicle is accelerated by a supersonic combustion ramjet having an inlet contraction ratio of 25. For a cruise Mach number of 10, the specific impulse and thrust coefficient at cruise were 2340 seconds and 0.93. Another form of the ERJ-SJ is the ejector dual-mode ramjet.

Propulsion System 5 (EDMRJ)

The ejector dual-mode ramjet is essentially an ejector ramjet and a SJ combined into a single engine. From takeoff to Mach 2, the engine operates as an augmented liquid oxygen - hydrogen rocket. The rocket operates at an O/F ratio of 7.94 while the secondary combustor operates at an equivalence ratio of 1.0. The ratio of the rocket propellant flow rate to inlet airflow was maintained at a value of 0.05 from takeoff to Mach 1. From Mach 1 to 2, this ratio was decreased linearly to zero. Between Mach 2 and 7, the engine operates as a subsonic combustion ramjet at an equivalence ratio of 1.0. Beyond Mach 7, the engine operates as a supersonic combustion ramjet. The inlet captured airflow schedule above Mach 4 was the same as for the dual-mode ramjet of system 2 which had an inlet contraction ratio of 10. Below Mach 4, some movable geometry was assumed so as to increase the capture mass flow ratio from 10 to 20 percent at Mach 1. This increase in the capture mass flow ratio was decreased linearly to zero at Mach 4. Figure 42 indicates the variation in specific impulse and thrust coefficient obtained with this system. For a cruise Mach number of 10, the thrust coefficient and specific impulse at cruise were 0.4 and 1850 seconds.

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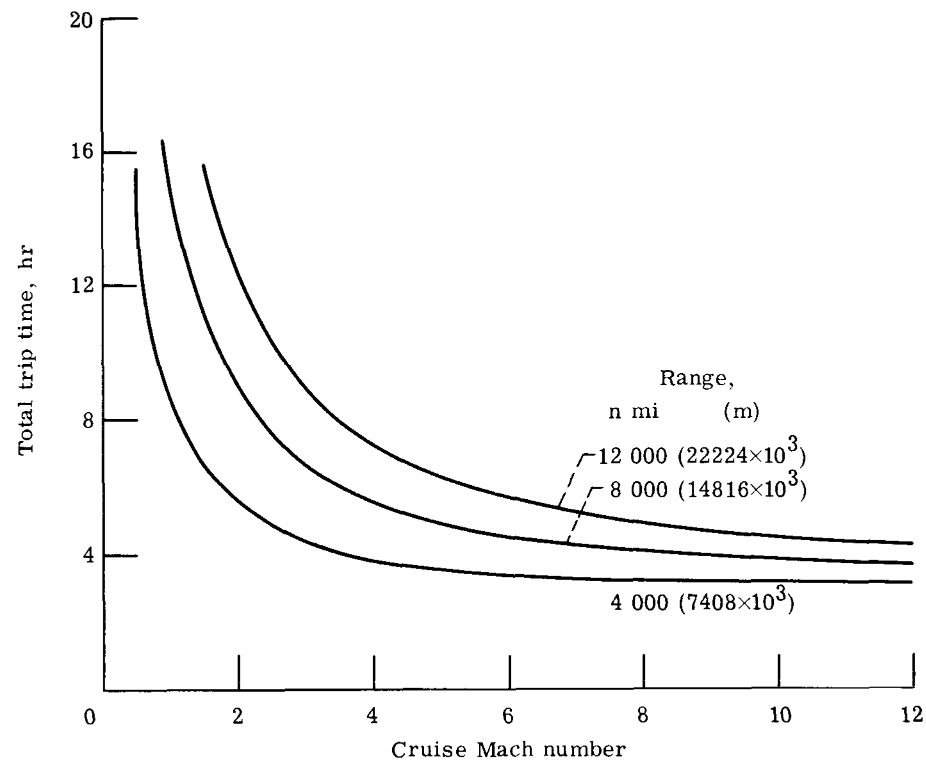
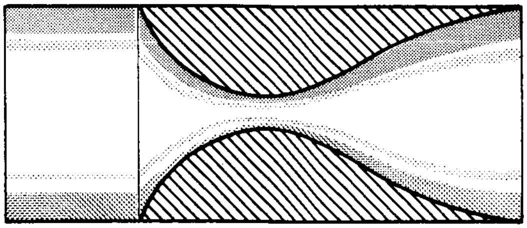
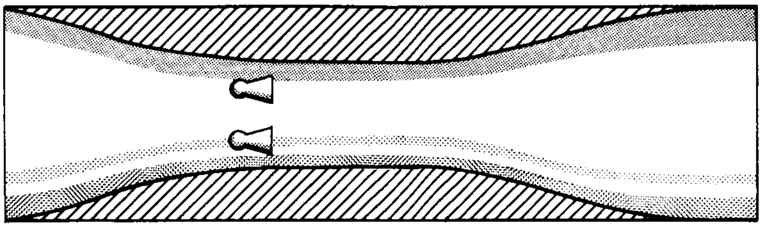


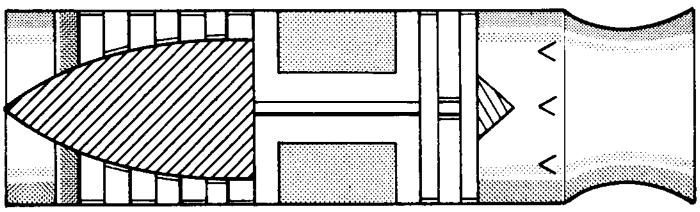
Figure 1. - Aircraft transit times. Ground transport, holds, etc., 2 hours; average acceleration, 0. 2 g.



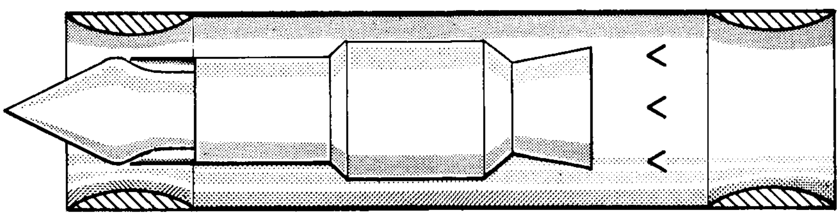
(a) Rocket (R).



(b) Ejector ramjet (ERJ).



(c) Turbojet (TJ).



(d) Turboramjet (TRJ).

Figure 2. - Initial acceleration engines.

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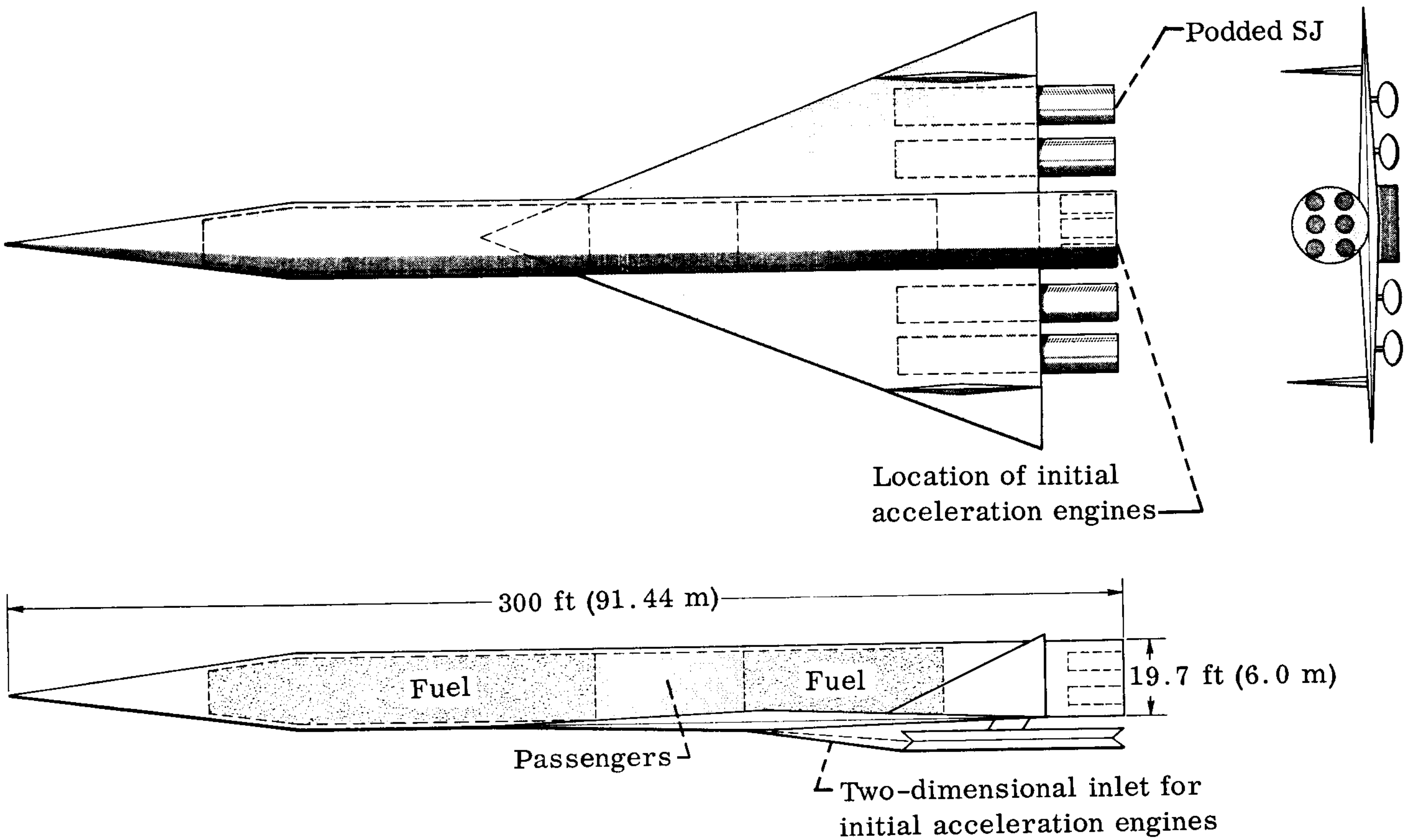

Figure 3. - Typical hypersonic transport configuration.

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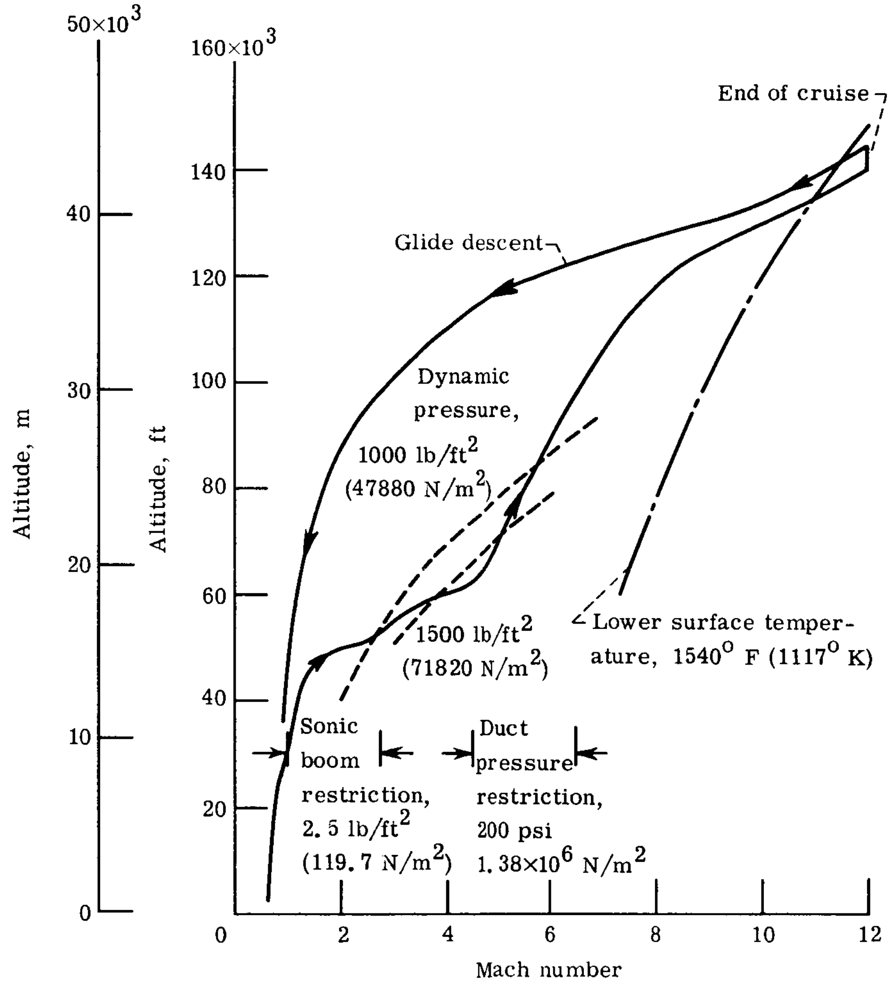
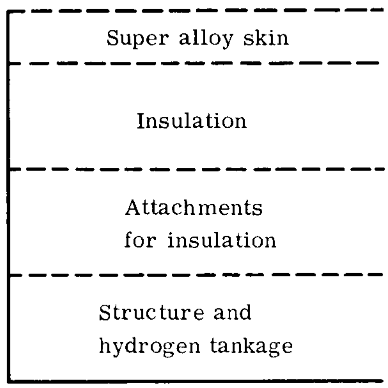
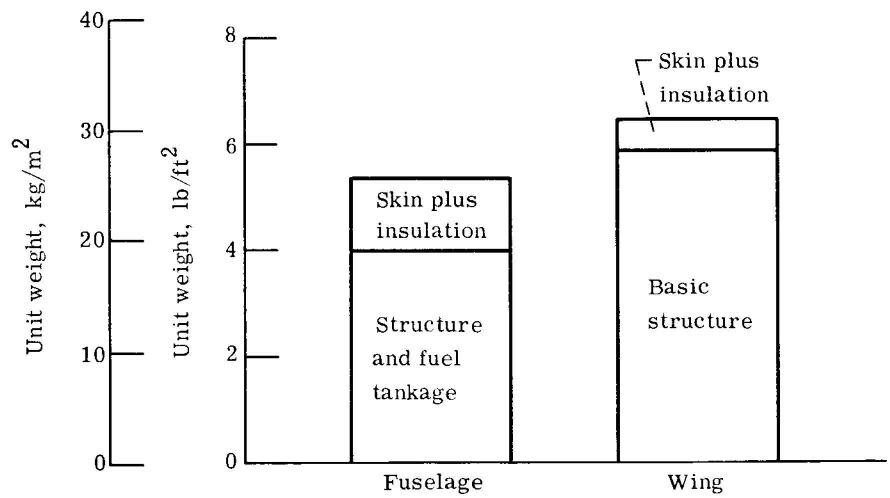


Figure 4. - Airplane flight path.

[redacted]

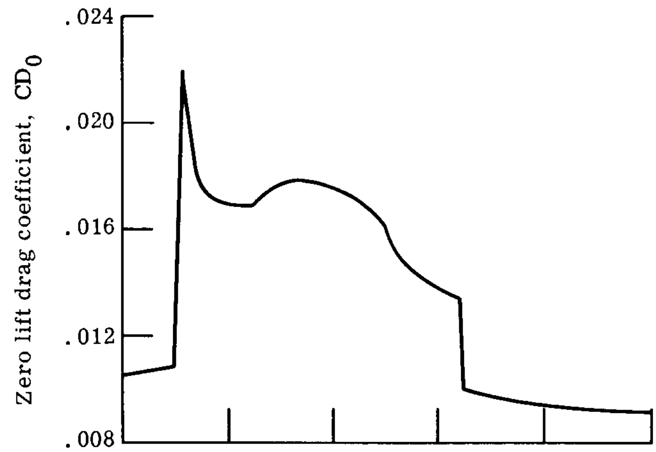


(a) Structural makeup.

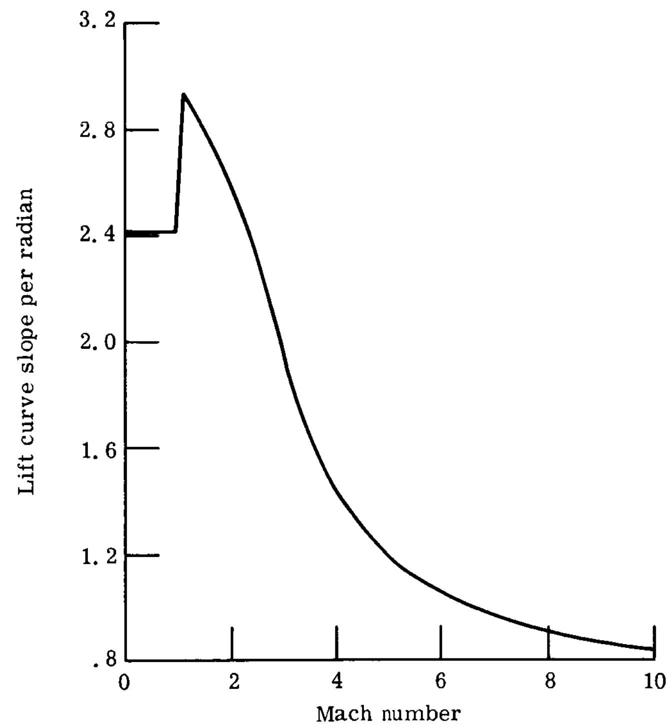


(b) Structural weight.

Figure 5. - Typical fuselage and wing structure.



(a) Zero lift drag.



(b) Lift curve slope.

Figure 6. - Airplane aerodynamic performance level.

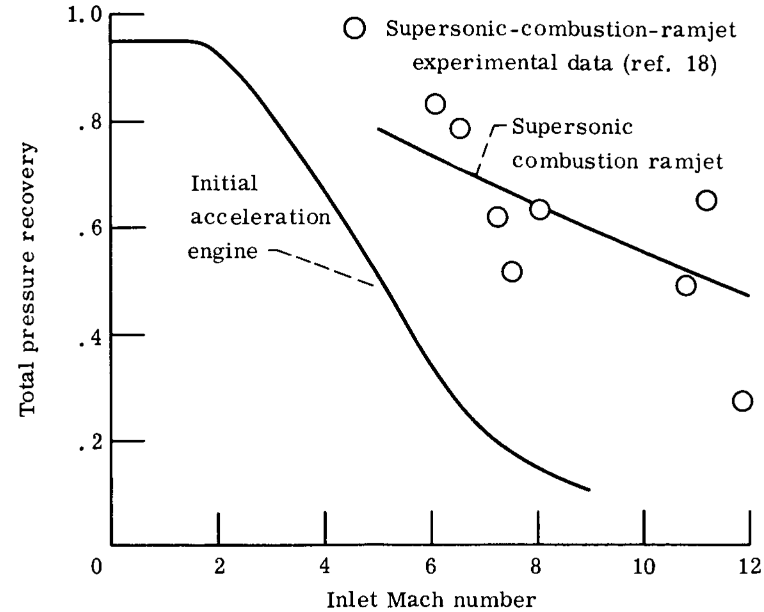


Figure 7. - Inlet pressure recovery for initial acceleration and supersonic combustion ramjet engines.

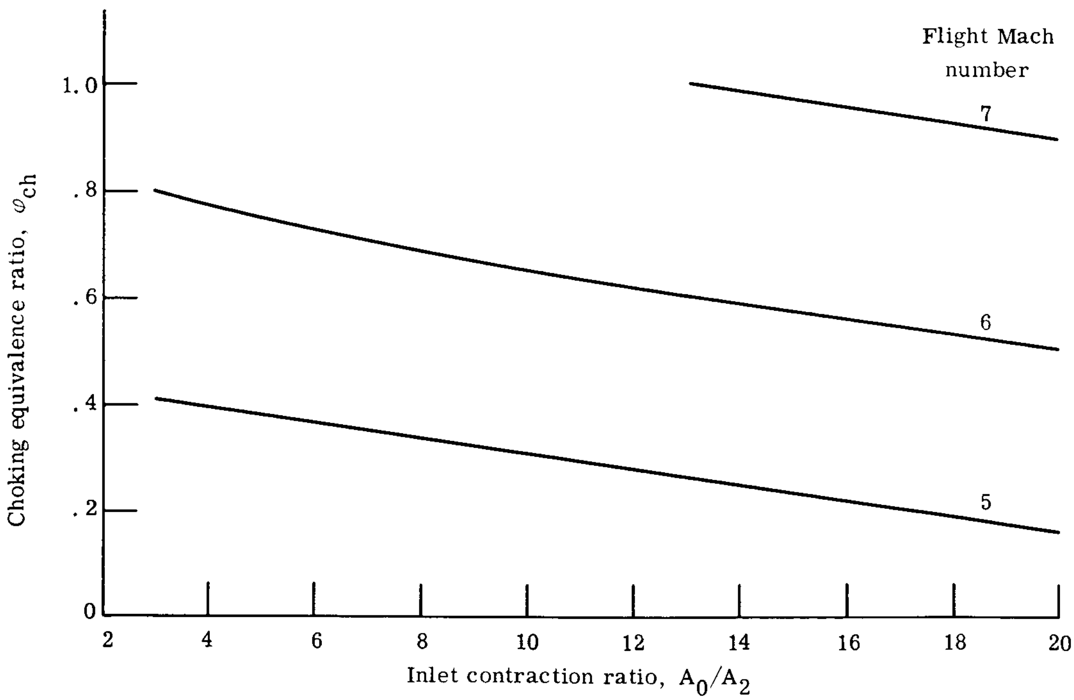


Figure 8. - Effect of inlet contraction ratio on choking equivalence ratio. Combustor area ratio, 1.35.

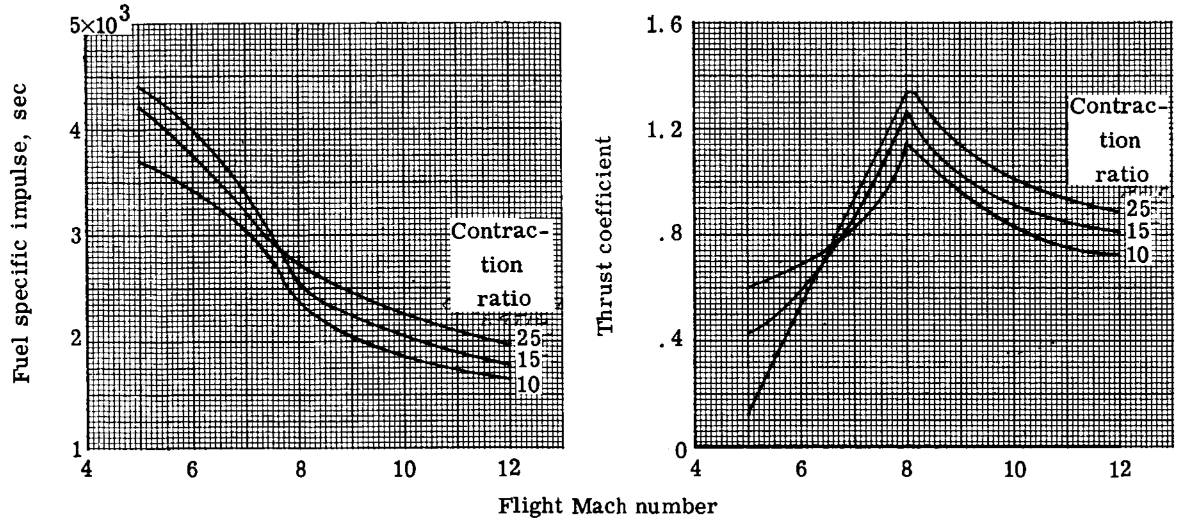


Figure 9. - Effect of inlet contraction ratio on supersonic-combustion-ramjet (SJ) specific impulse and thrust coefficient.

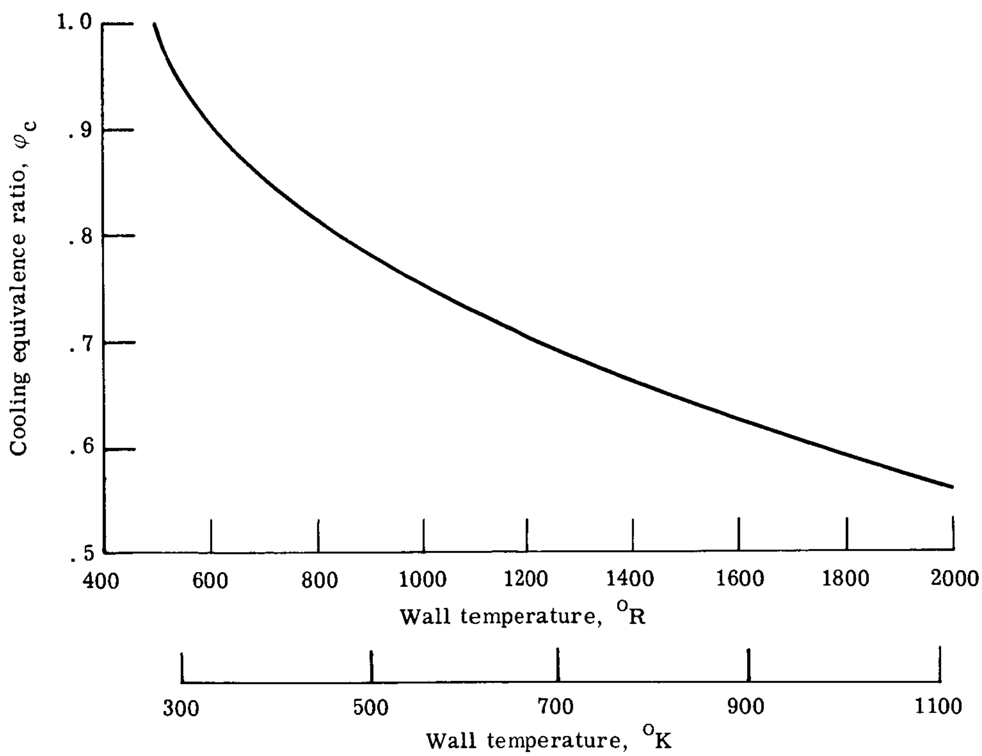


Figure 10. - Cooling equivalence ratios required to maintain various wall temperatures for the supersonic combustion ramjet. Cruise Mach number, 10; inlet contraction ratio, 10.

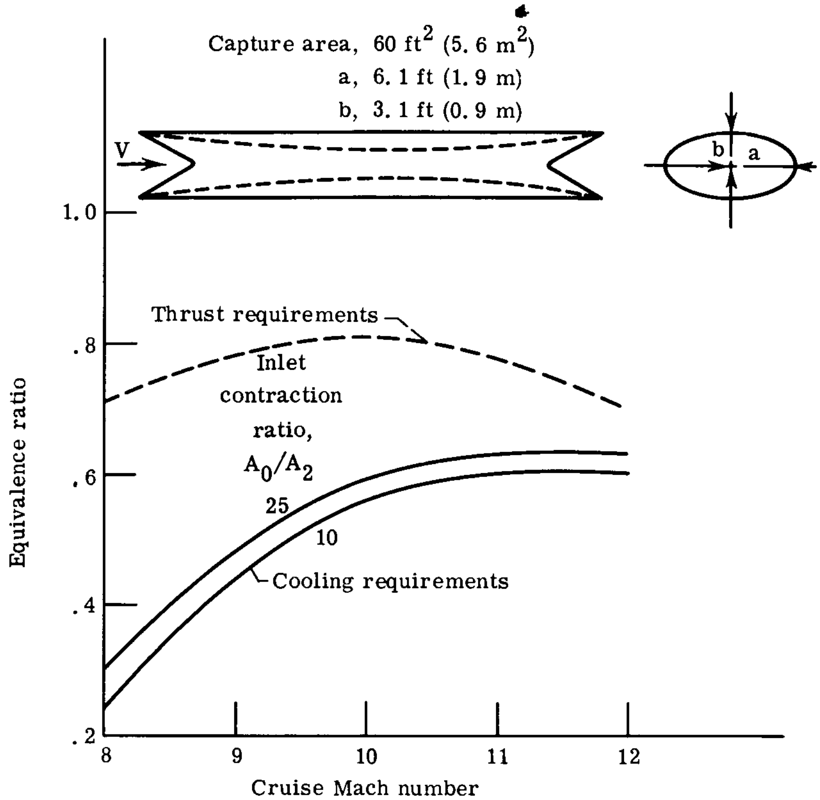


Figure 11. - Comparison of equivalence ratio required for engine internal cooling with thrust requirements. Wall temperature, 2000° R.

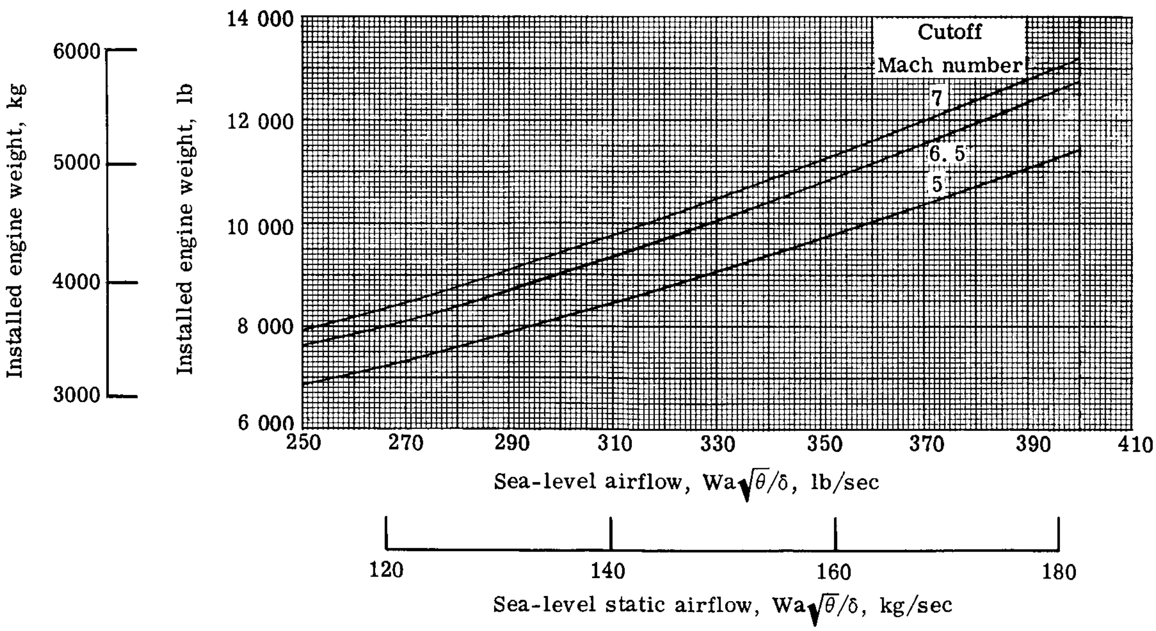


Figure 12. - Installed turboramjet weight.

[redacted]

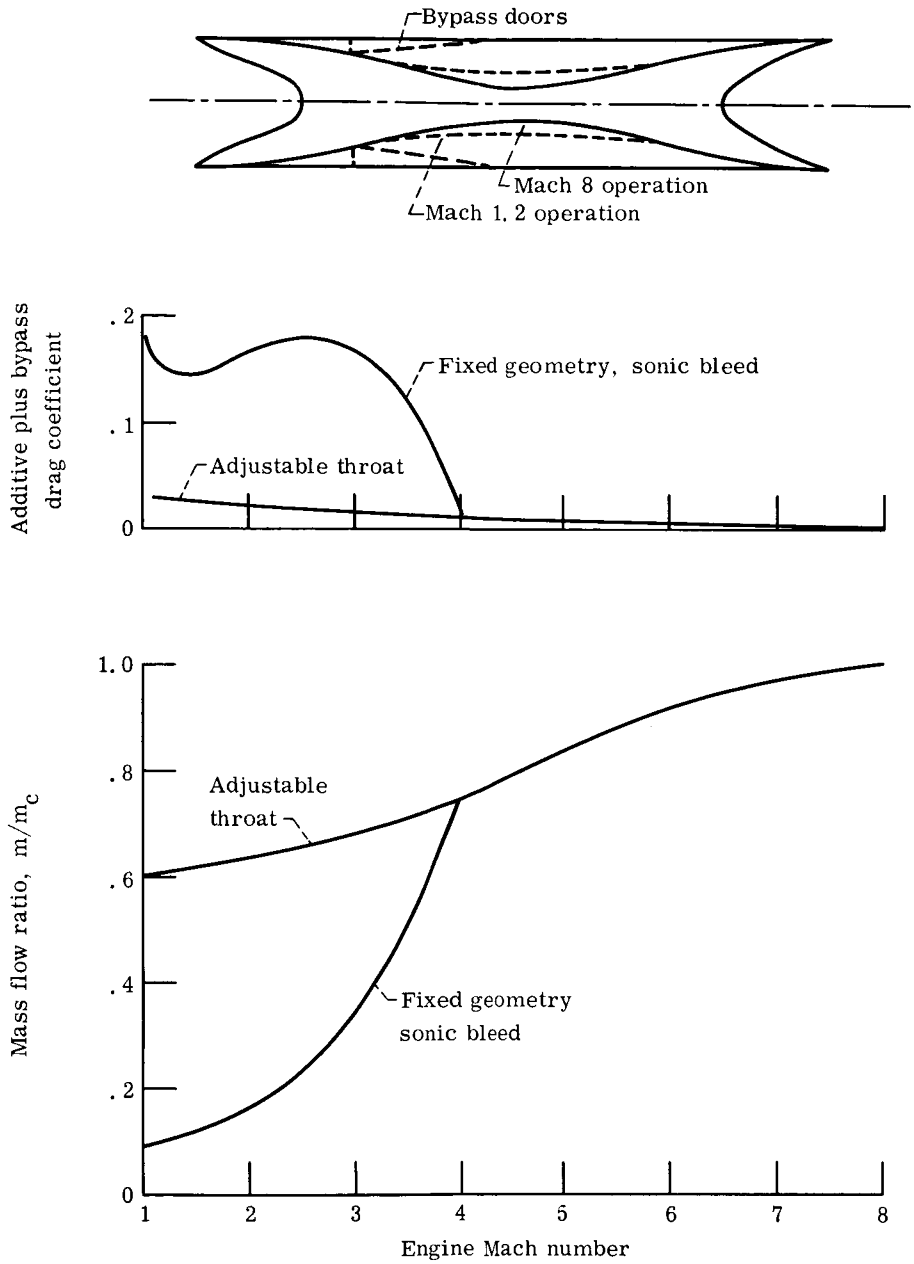


Figure 13. - Dual-mode ramjet inlet characteristics.

[redacted]

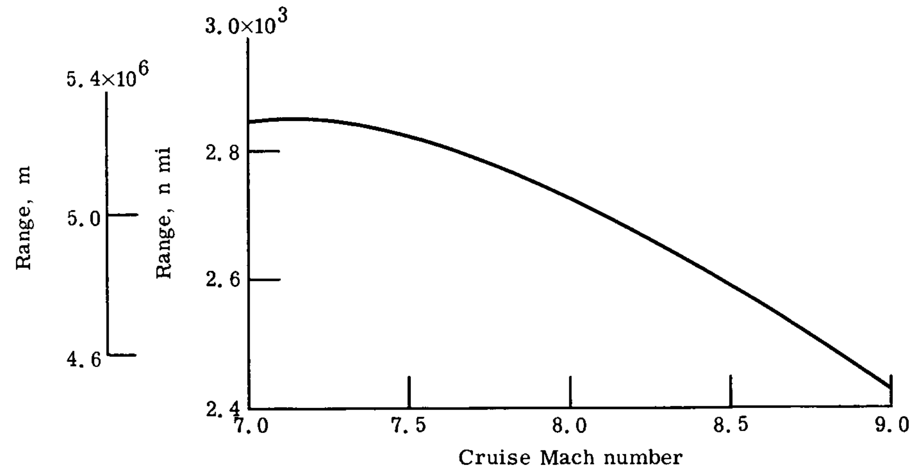


Figure 14. - Effect of cruise Mach number on airplane range to end of cruise. Propulsion system 1 (TJ-R-SJ).

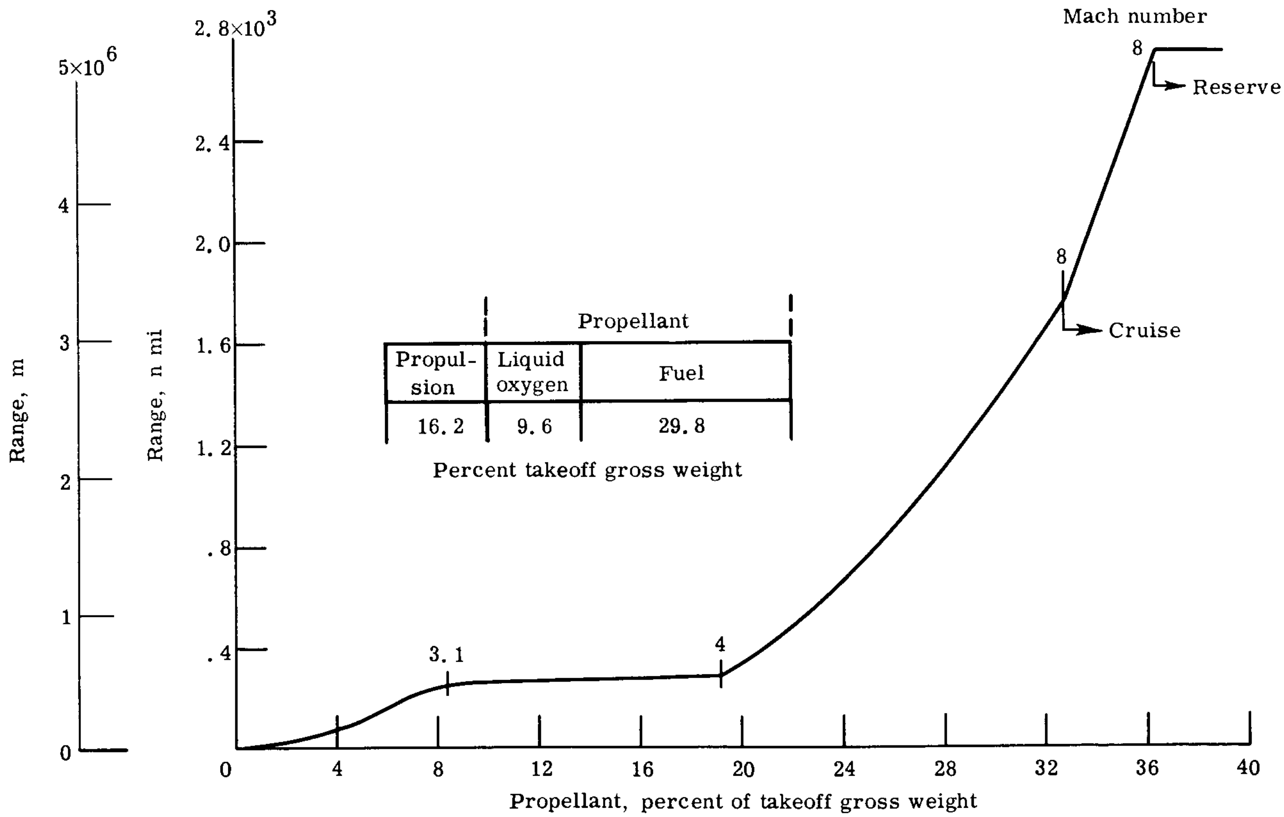


Figure 15. - Range and propellant fraction to end of cruise. Propulsion system 1 (TJ-R-SJ).

[redacted]

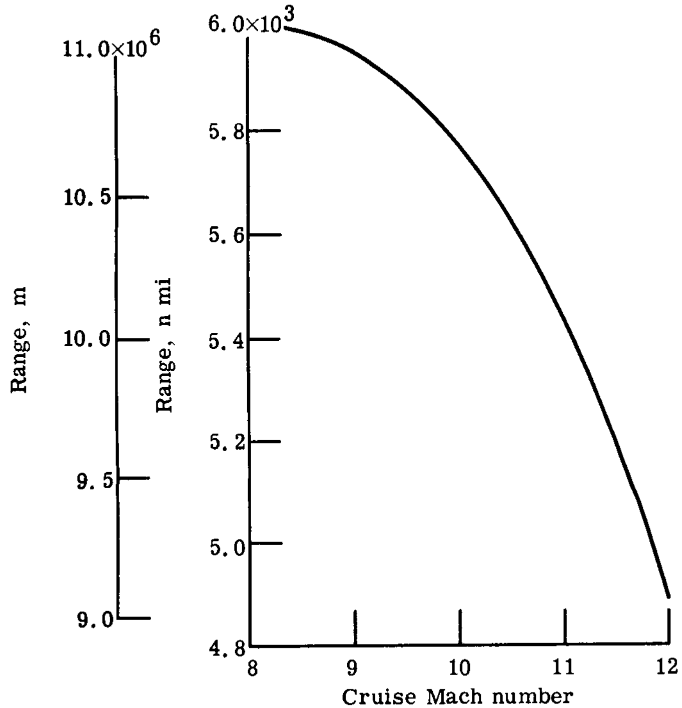
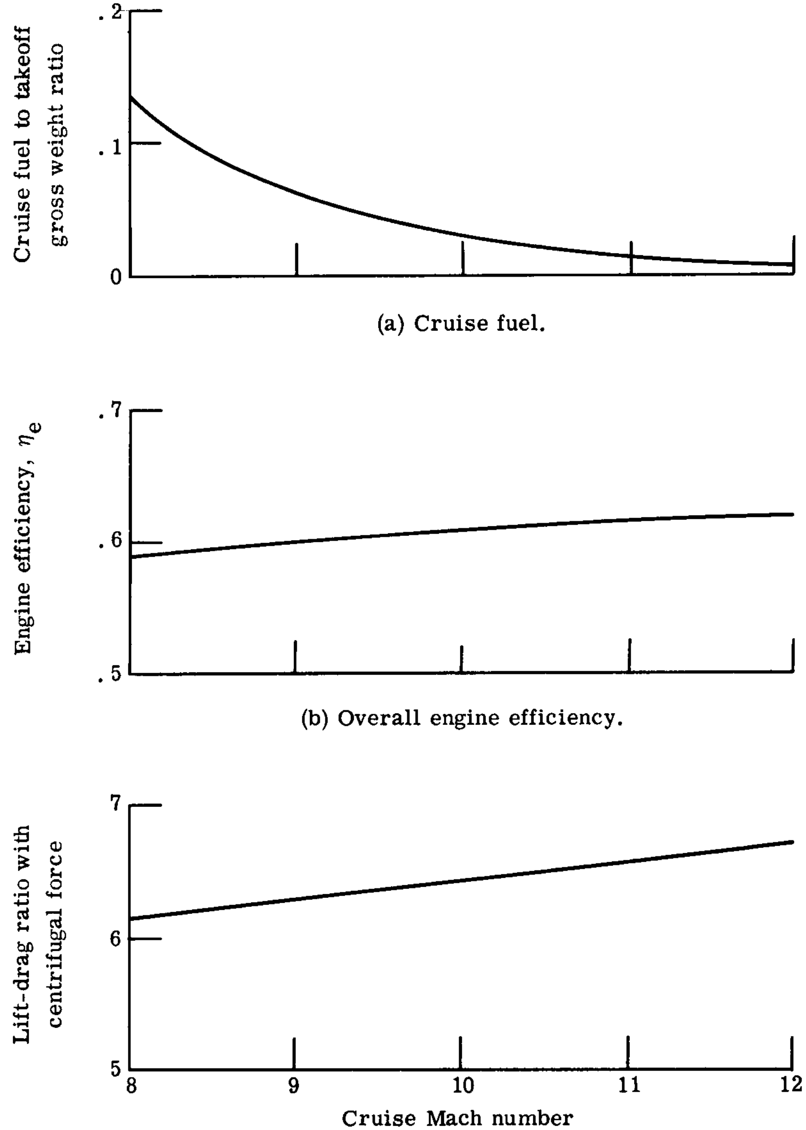


Figure 16. - Effect of cruise Mach number on airplane range to end of cruise. Propulsion system 2 (TRJ-SJ).



(a) Cruise fuel.

(b) Overall engine efficiency.

(c) Aerodynamic efficiency.

Figure 17. - Airplane performance levels. Propulsion system 2 (TRJ-SJ).

[redacted]

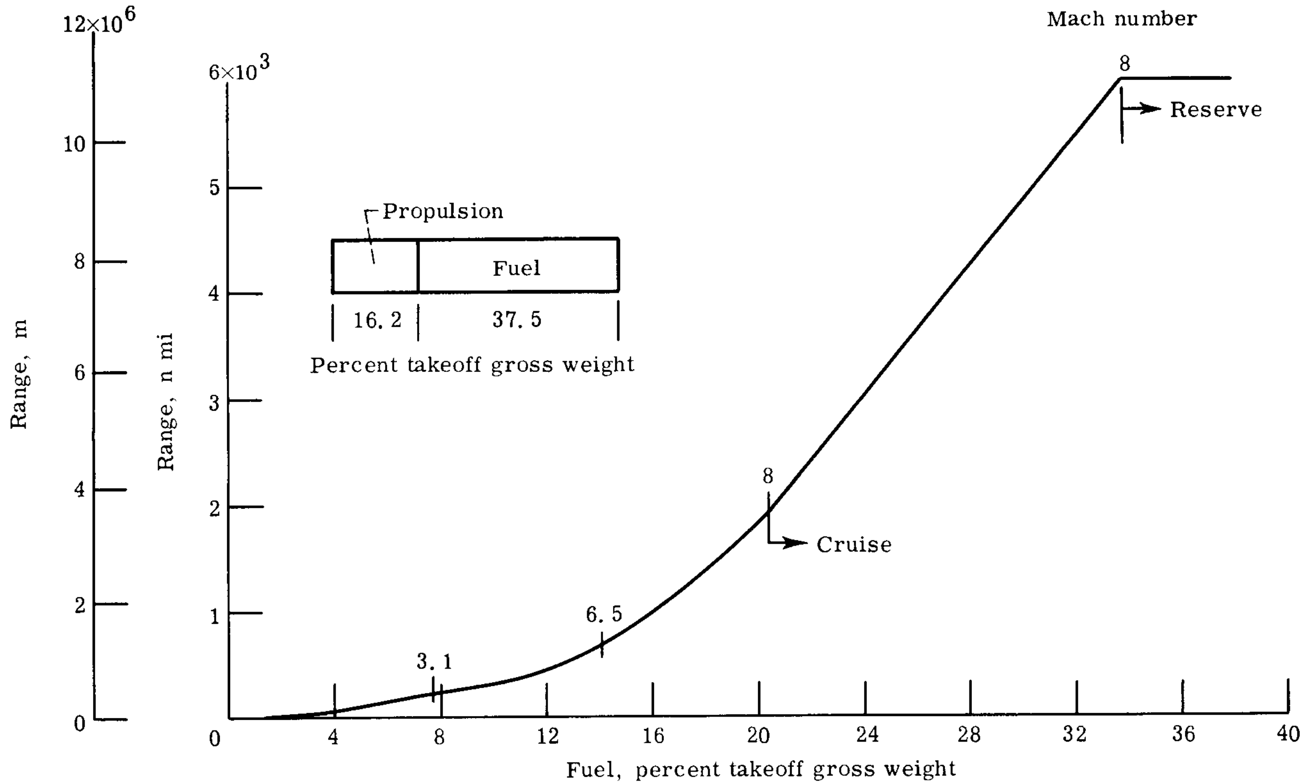


Figure 18. - Range and fuel fraction to end of cruise. Propulsion system 2 (TRJ-SJ).

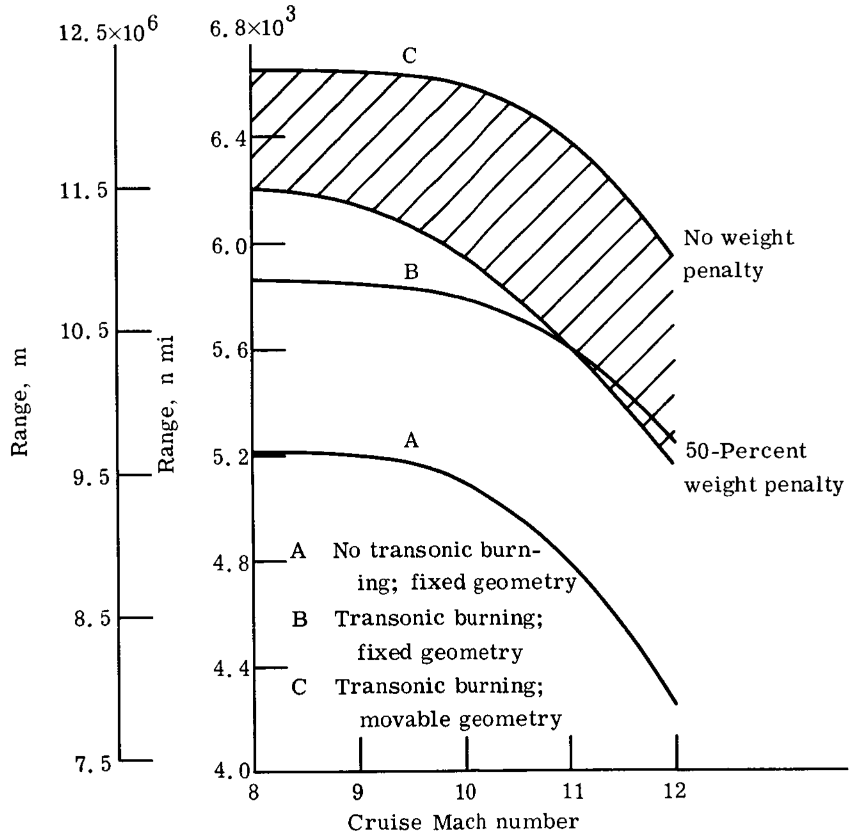


Figure 19. - Range to end of cruise. Propulsion system 3 (TJ-DMRJ); inlet contraction ratio, 10; transonic burning-ramjet augmentation from Mach 1. 2 to 3. 1.

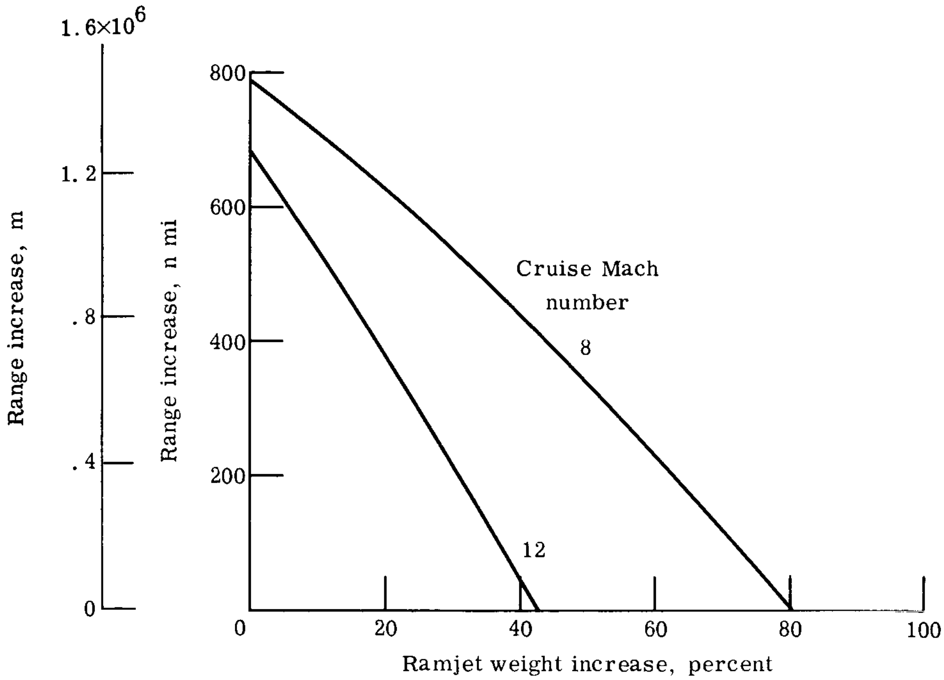


Figure 20. - Increase attained in range to end of cruise with scheme C relative to scheme B when DMRJ weight is penalized for movable geometry feature. Propulsion system 3 (TJ-DMRT).

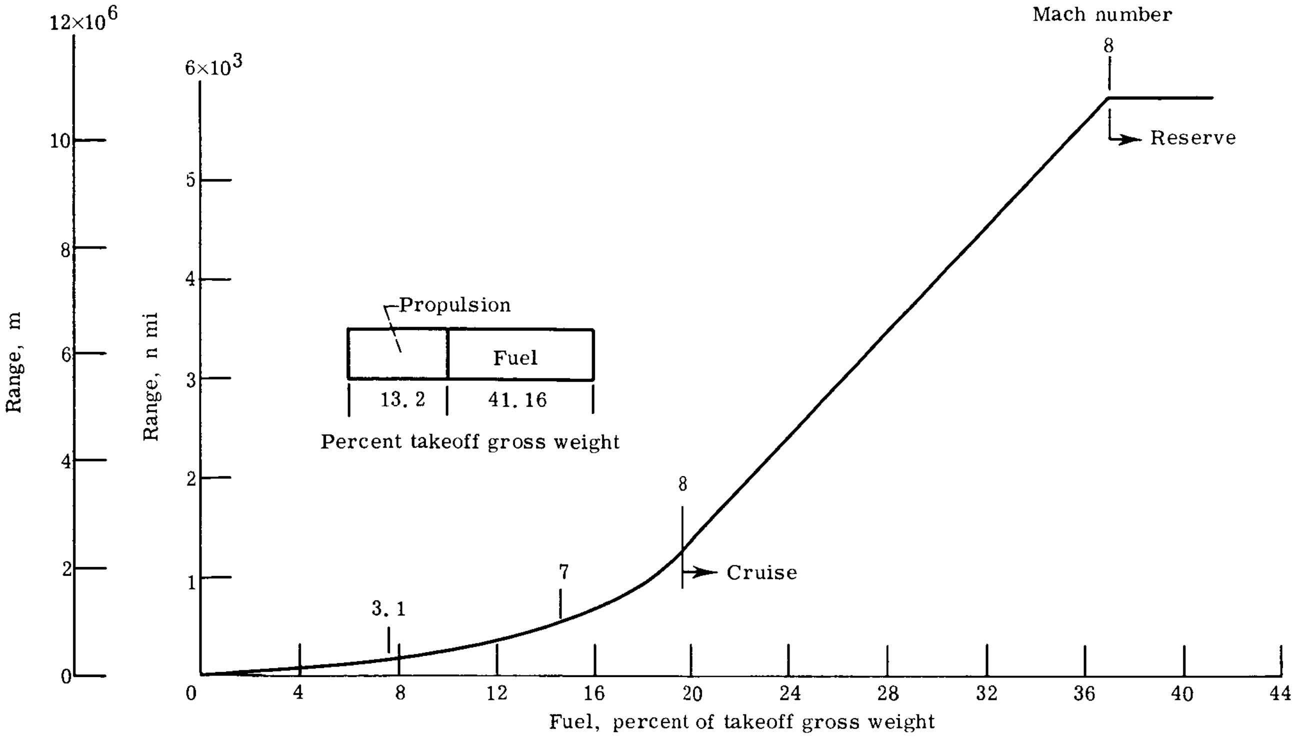


Figure 21. - Range and fuel fraction to end of cruise. Propulsion system 3 (TJ-DMRJ).

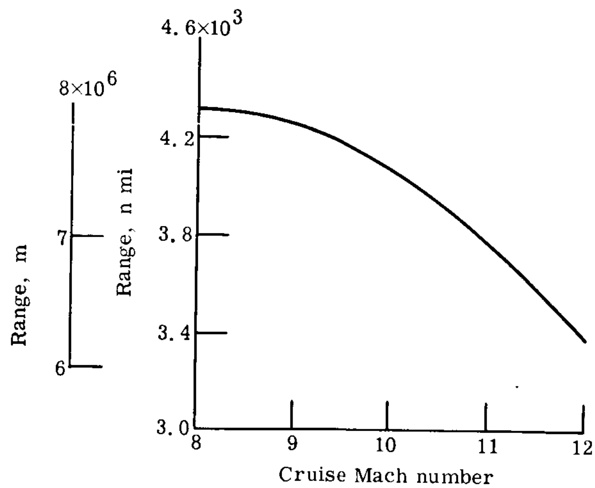


Figure 22. - Effect of cruise Mach number on airplane range to end of cruise. Propulsion system 4 (ERJ-SJ).

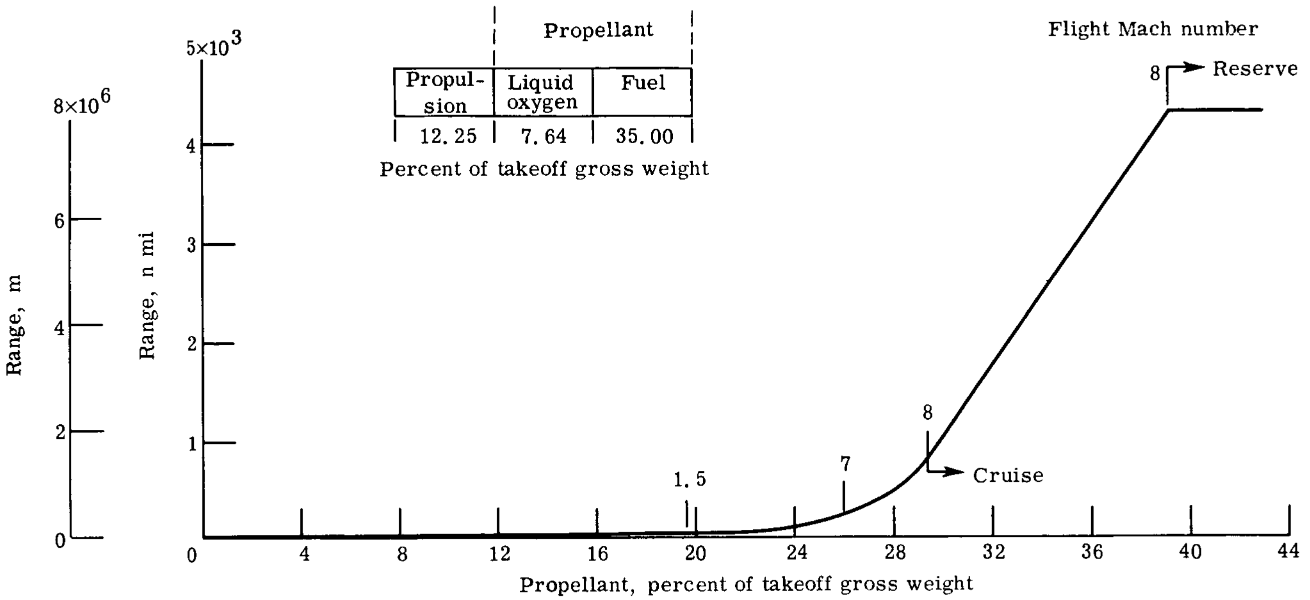


Figure 23. - Range and propellant fraction to end of cruise. Propulsion system 4 (ERJ-SJ).

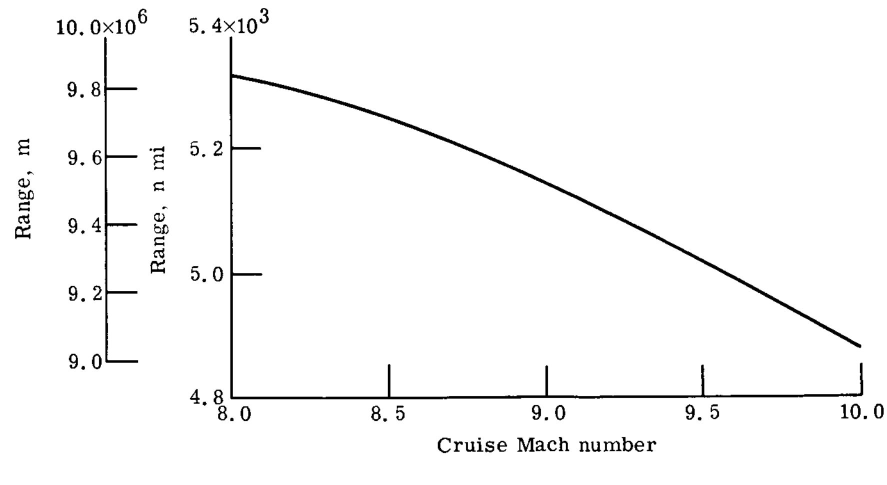


Figure 24. - Effect of cruise Mach number on airplane range to end of cruise. Propulsion system 5 (EDMRJ).

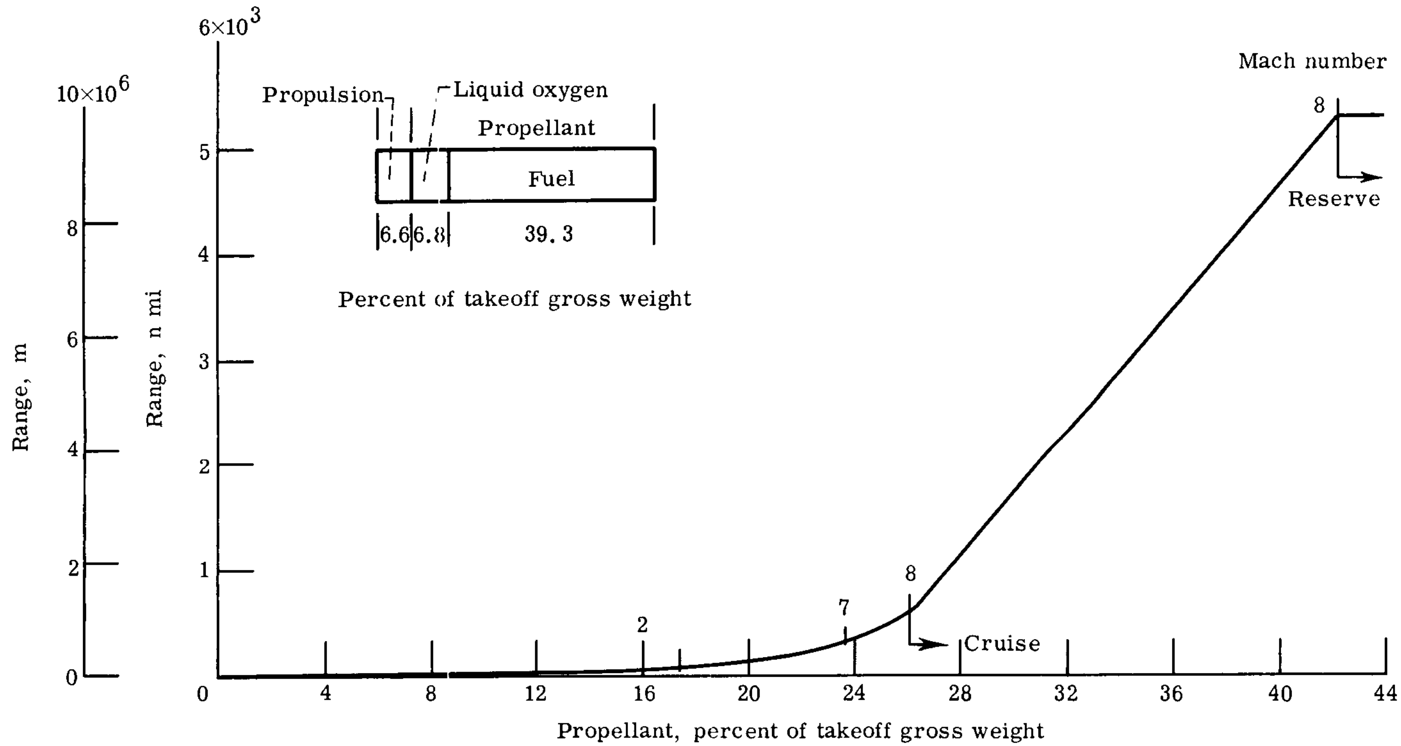


Figure 25. - Range and propellant fraction to end of cruise. Propulsion system 5 (EDMRJ).

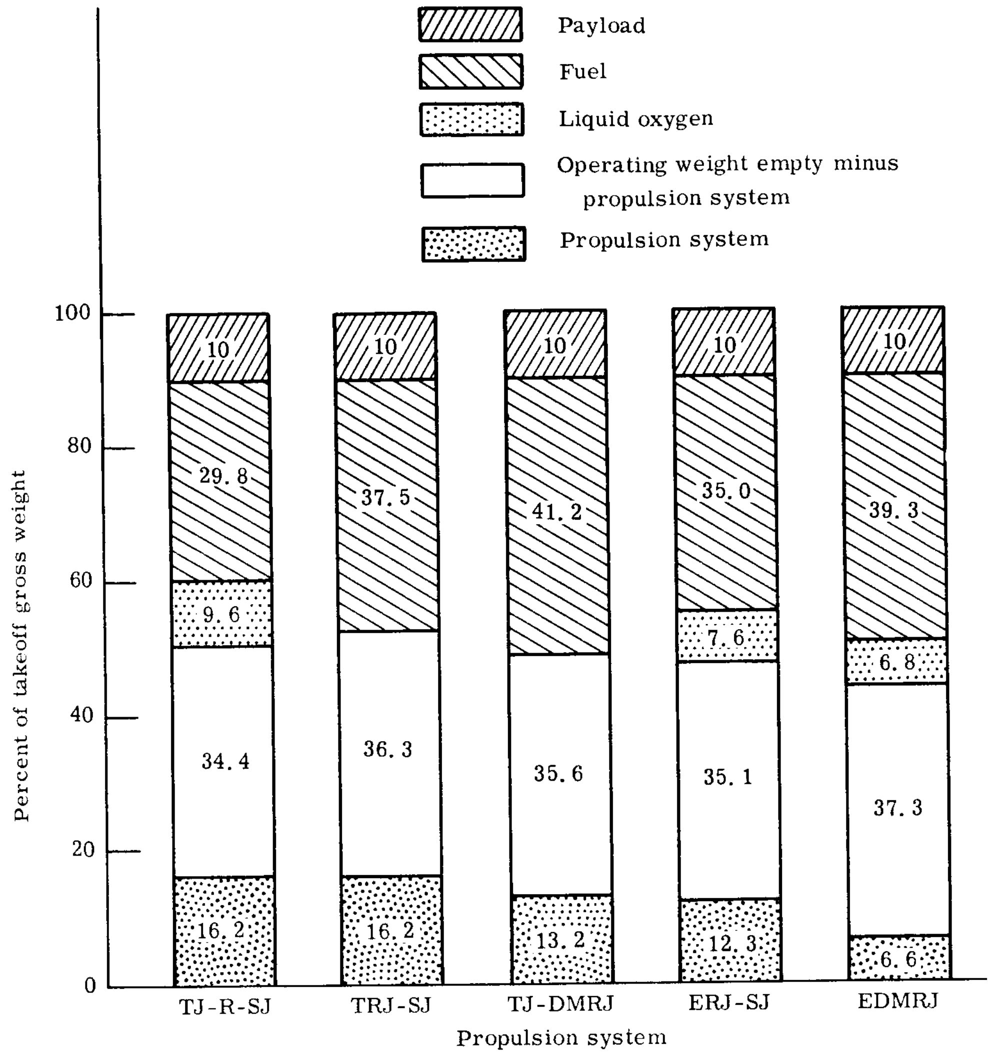
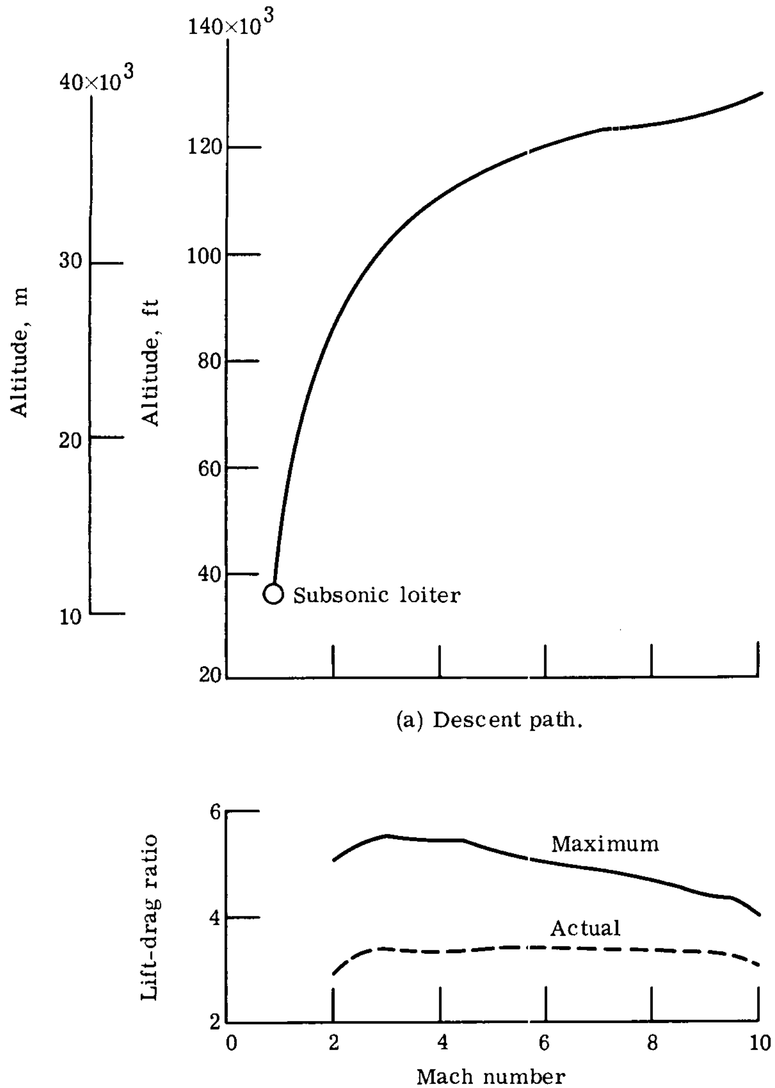


Figure 26. - Airplane weight breakdown for five propulsion systems. Cruise Mach number, 8.

[redacted]



(a) Descent path.

(b) Lift-drag ratio.

Figure 27. - Airplane descent flight path and typical lift-drag ratio variation. Propulsion system 2 (TRJ-SJ).

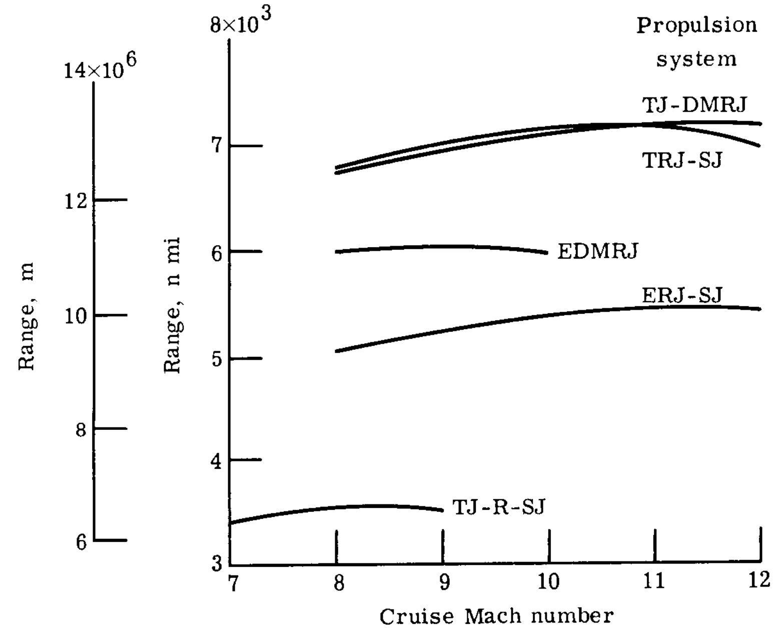


Figure 28. - Comparison of range attained to end of descent with various propulsion systems.

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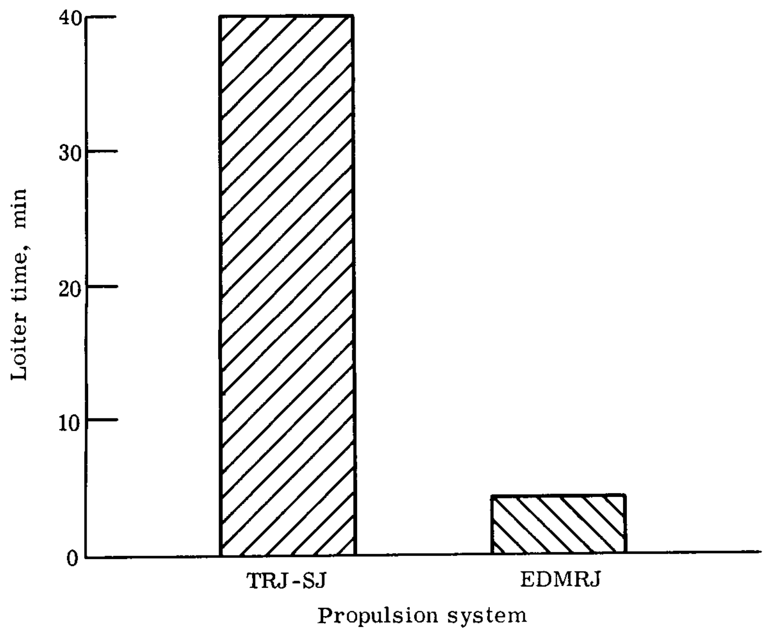
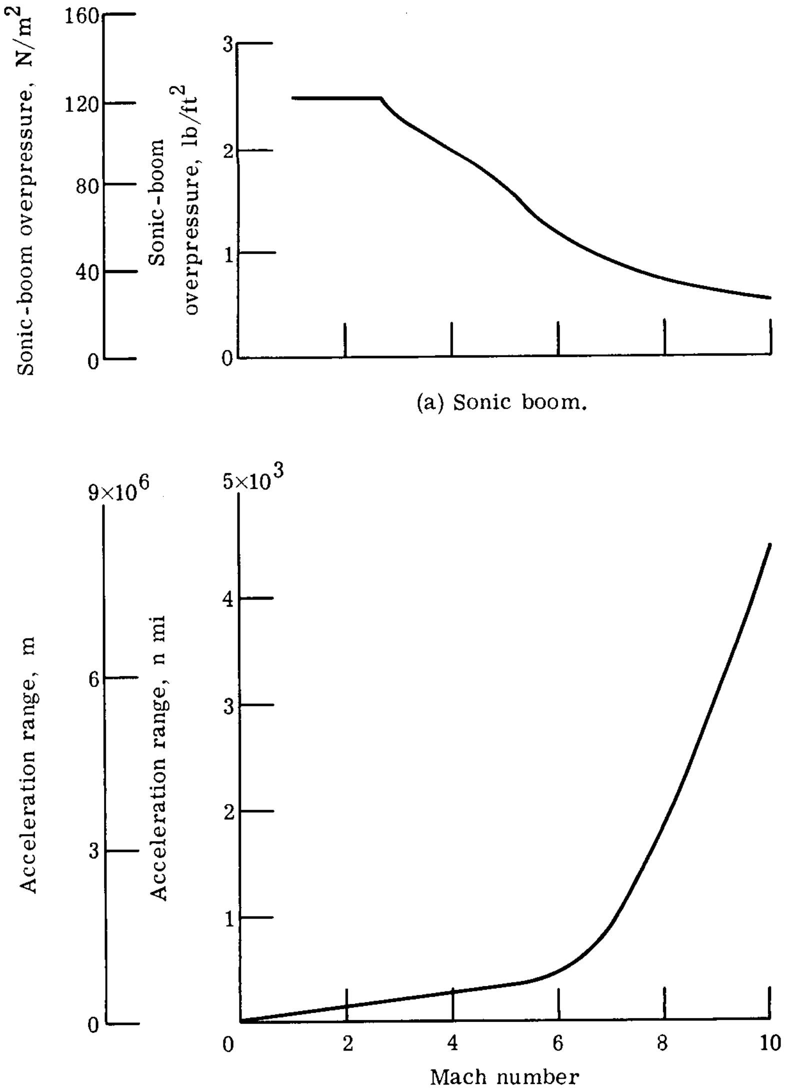


Figure 29. - Effect of propulsion system on subsonic loiter time. Cruise Mach number, 10; loiter Mach number, 0.9; altitude, 36 000 feet (10 973 m).



(a) Sonic boom.

(b) Acceleration range.

Figure 30. - Sonic-boom overpressure during acceleration phase of flight. Propulsion system 2 (TRJ-SJ).

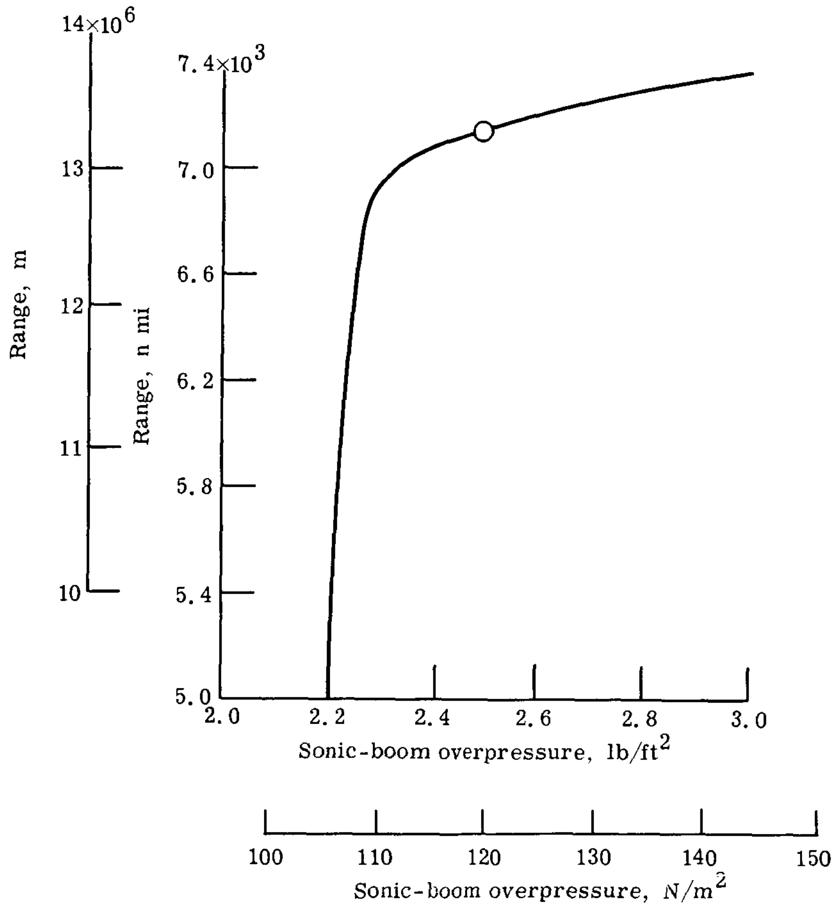
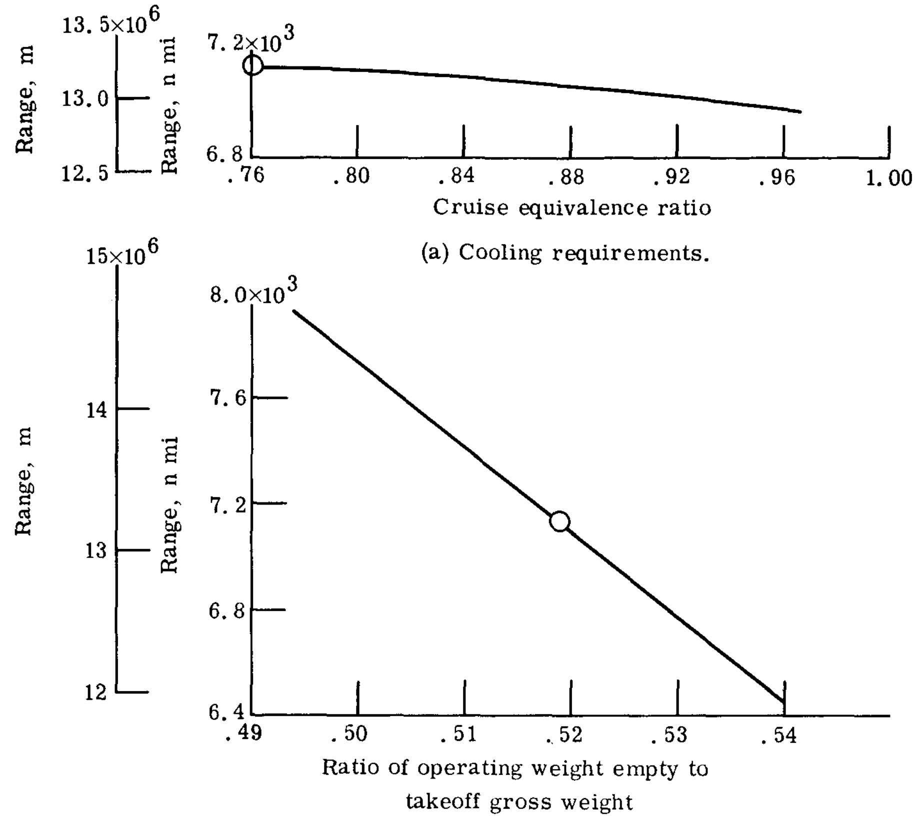


Figure 31. - Effect of transonic sonic-boom overpressure on range to end of descent. Propulsion system 2 (TRJ-SJ); cruise Mach number, 10.

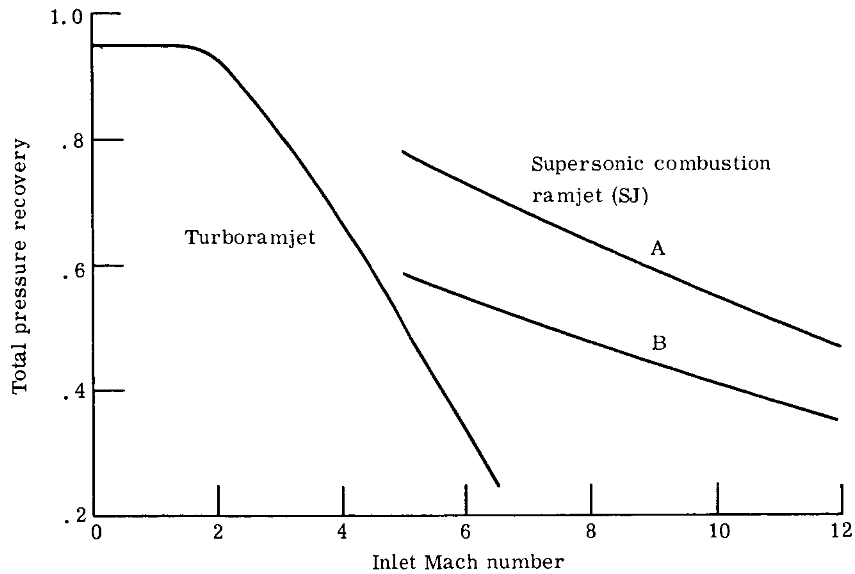


(a) Cooling requirements.

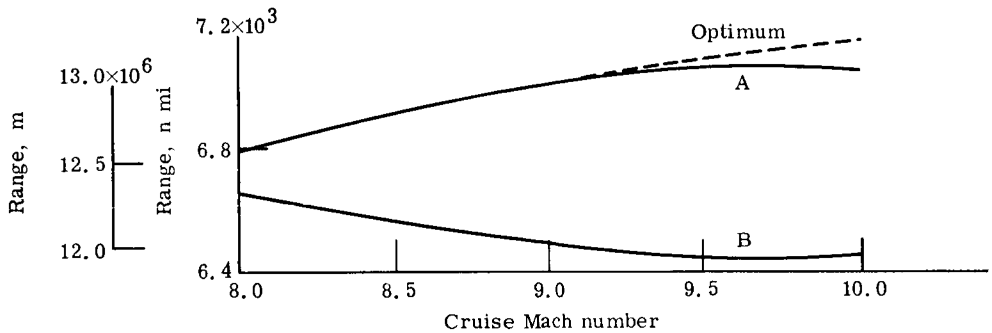
(b) Operating weight empty.

Figure 32. - Effect of supersonic-combustion-ramjet (SJ) engine cooling requirements and vehicle operating weight empty on range to end of descent. Propulsion system 2 (TRJ-SJ); cruise Mach number, 10.

[redacted]



(a) Inlet pressure recovery schedule.



(b) Airplane range.

Figure 33. - Effect of supersonic-combustion-ramjet (SJ) inlet pressure recovery on airplane range to end of descent. Propulsion system 2 (TRJ-SJ).

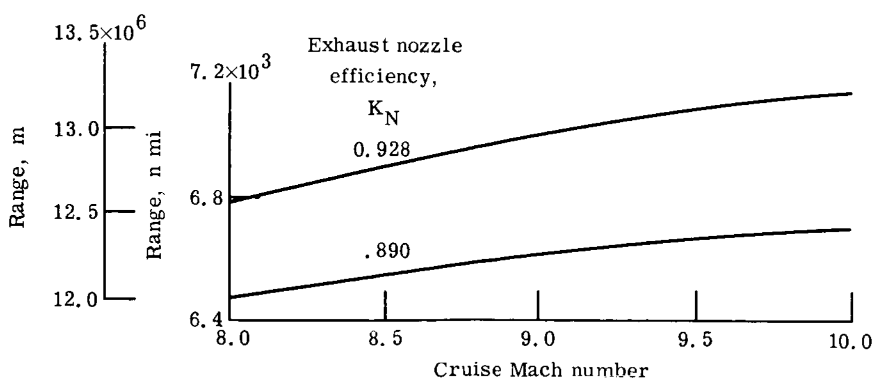


Figure 34. - Effect of supersonic-combustion-ramjet exhaust nozzle efficiency on airplane range to end of descent. Propulsion system 2 (TRJ-SJ).

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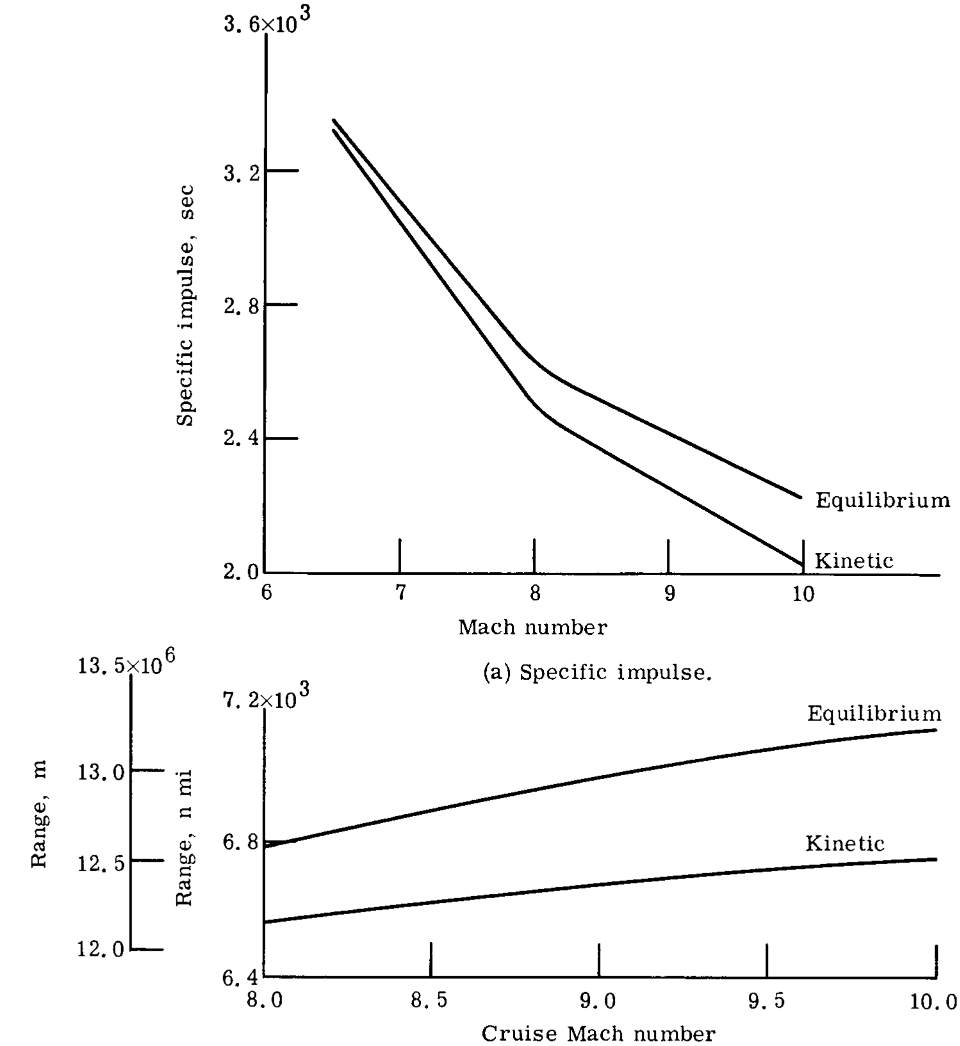


(a) Specific impulse.

(b) Range.

Figure 35. - Supersonic-combustion-ramjet nozzle kinetic effects on engine performance and range to end of descent. Propulsion system 2 (TRJ-SJ).

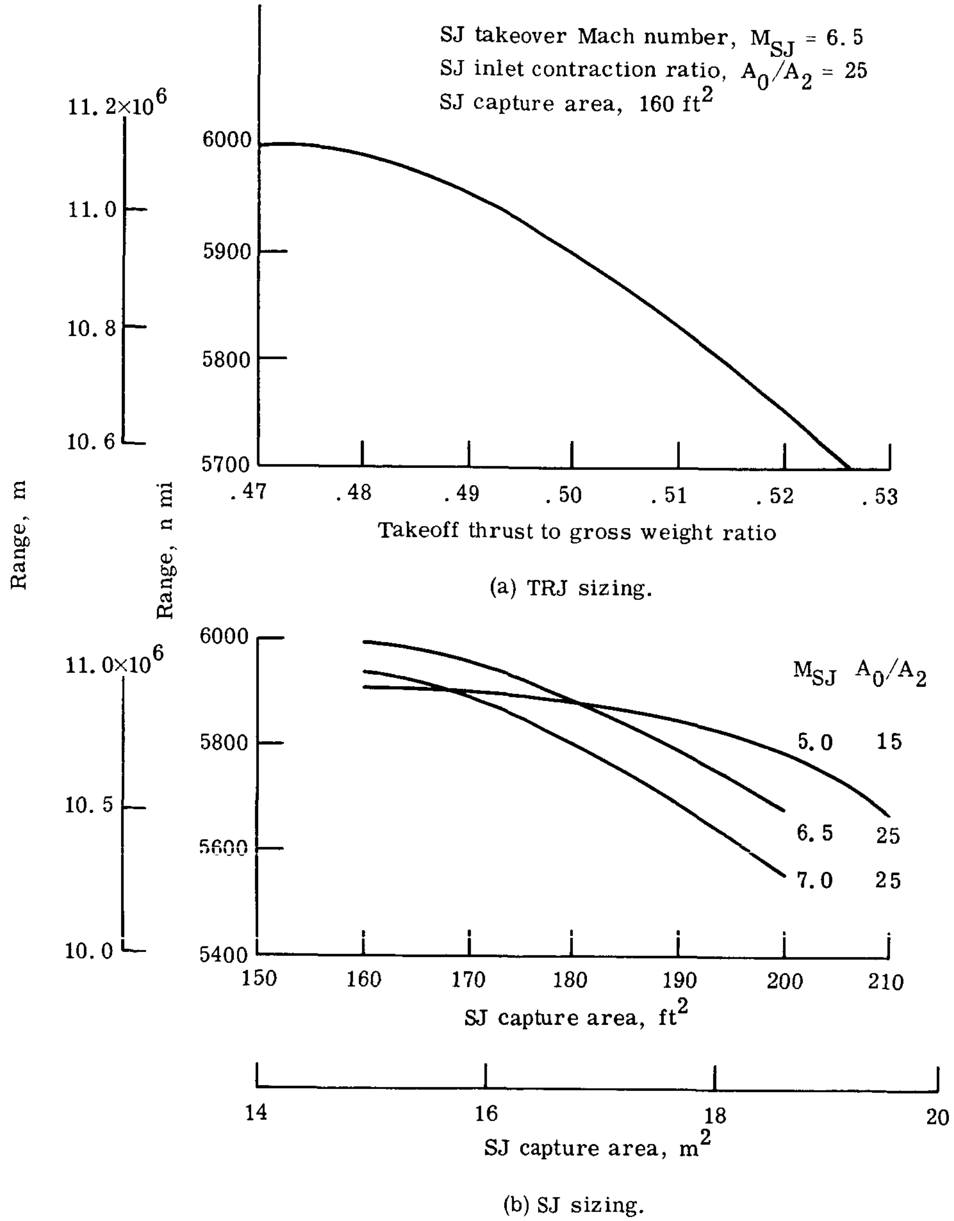
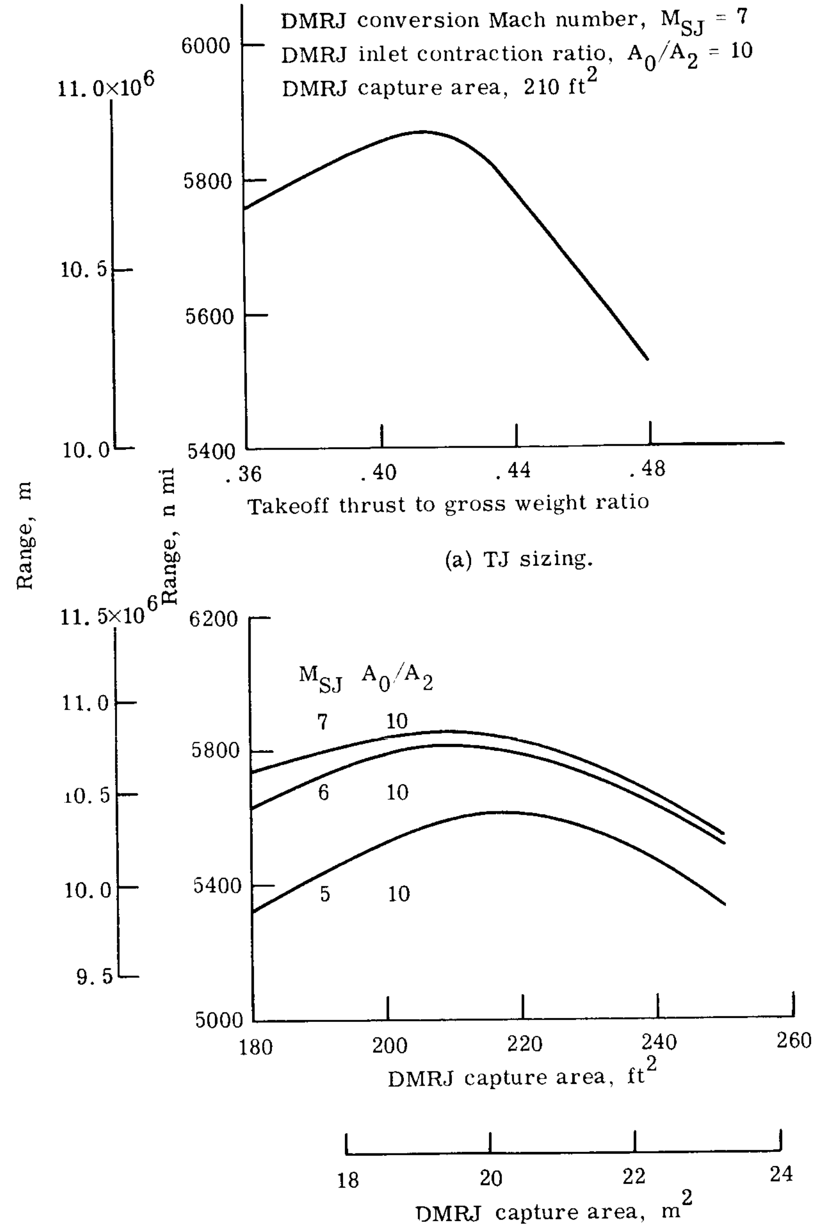


Figure 36. - Engine sizing for propulsion system 2 (TRJ-SJ) as function of range to end of cruise. Cruise Mach number, 8.

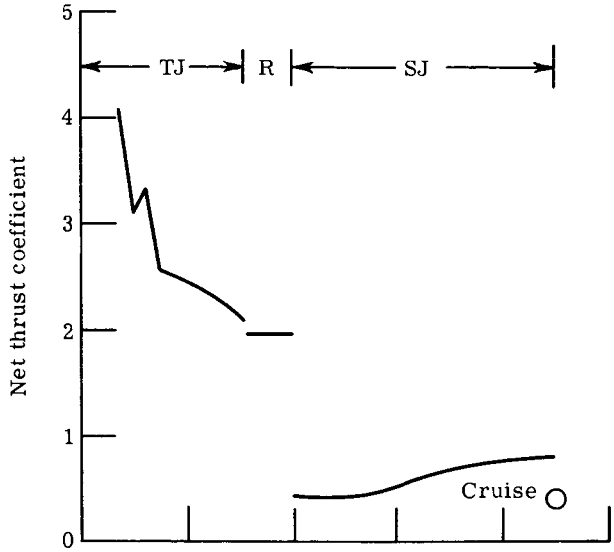


(a) TJ sizing.

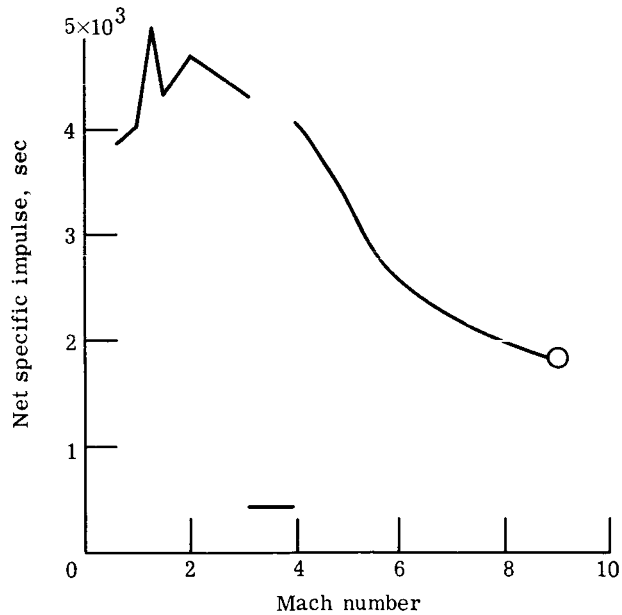
(b) SJ sizing.

Figure 37. - Engine sizing for propulsion system 3 (TJ-DMRJ) as function of range to end of cruise. Cruise Mach number, 8.

[redacted]



(a) Thrust coefficient.



(b) Specific impulse.

Figure 38. - Performance levels for propulsion system 1 (TJ-R-SJ).

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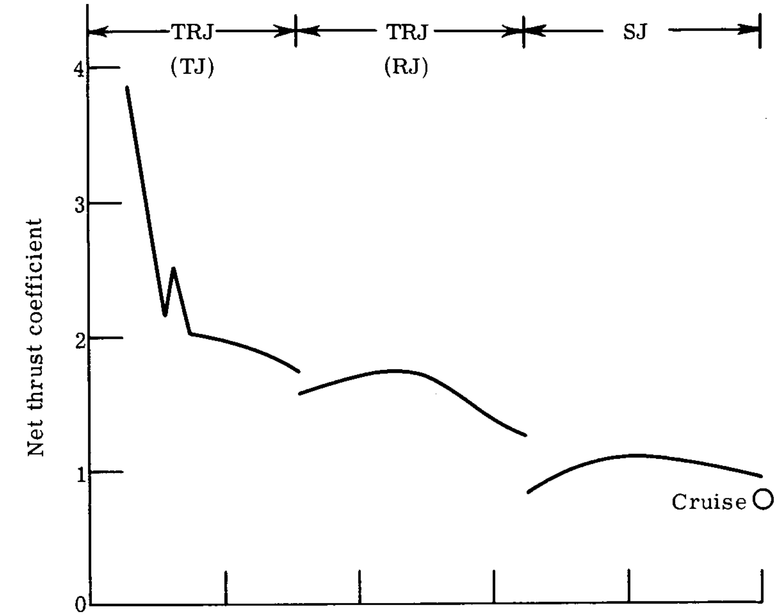
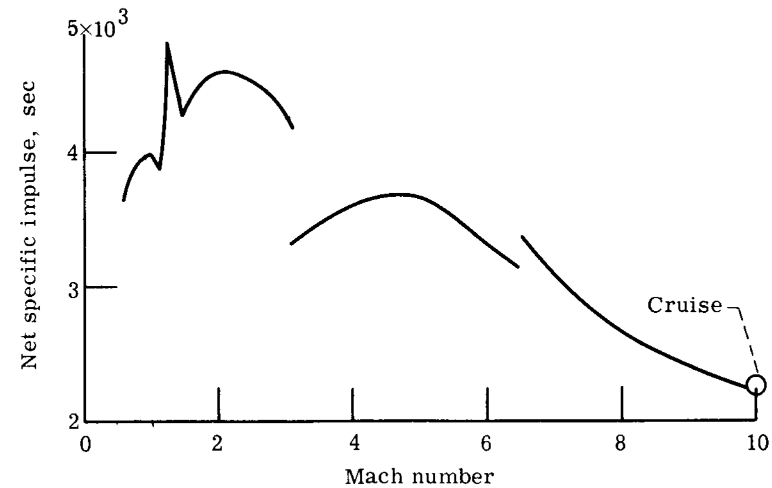
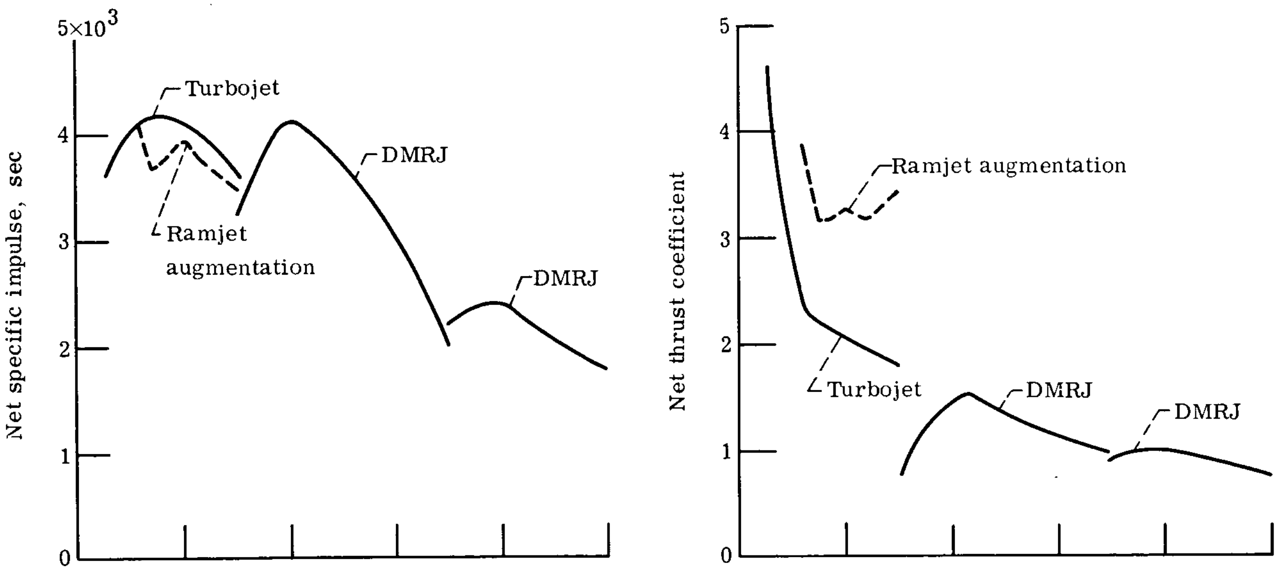


(a) Thrust coefficient.

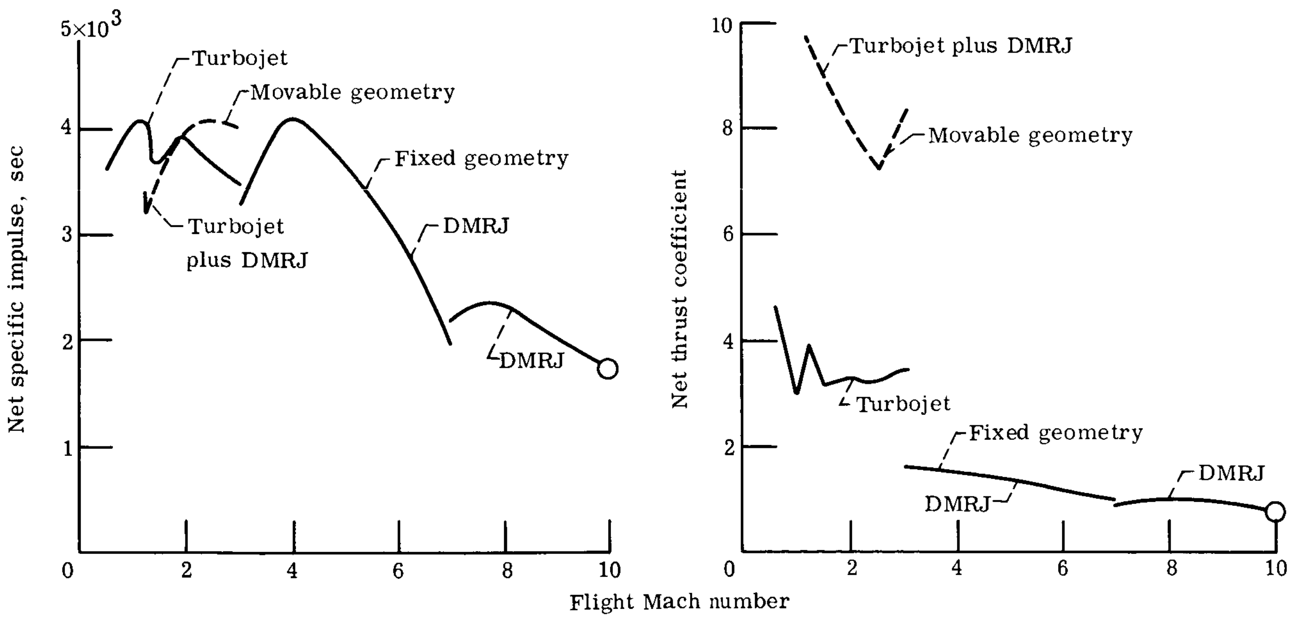


(b) Specific impulse.

Figure 39. - Performance levels for propulsion system 2 (TRJ-SJ).



(a) Fixed geometry for DMRJ.



(b) Movable geometry for DMRJ.

Figure 40. - Performance levels for propulsion system 3 (TJ-DMRJ). Mach numbers for ramjet augmentation, 1.2 to 3.1.

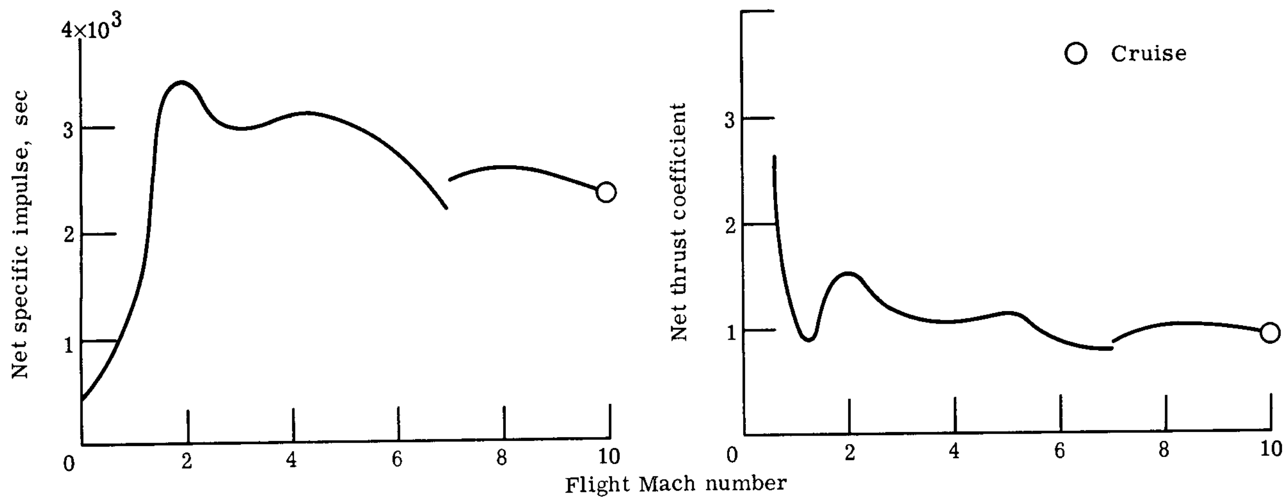
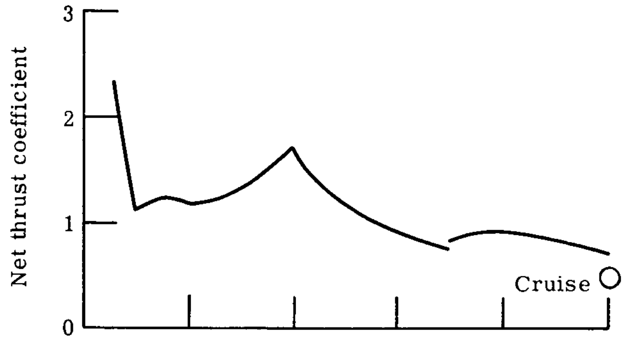
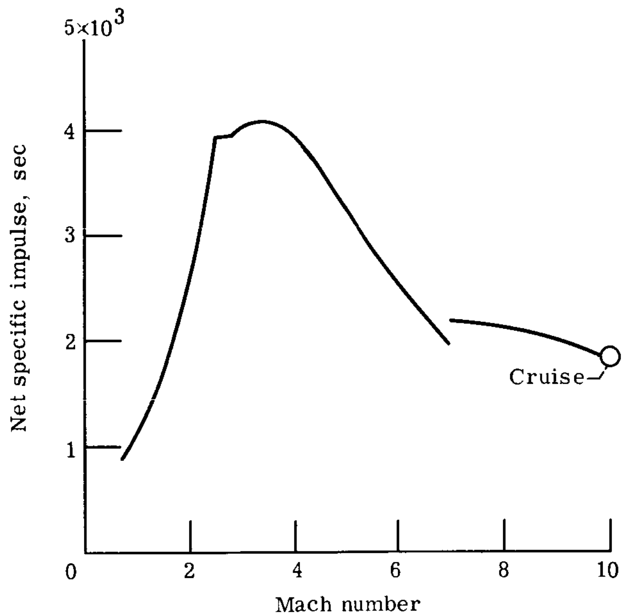


Figure 41. - Performance levels for propulsion system 4 (ERJ-SJ).



(a) Thrust coefficient.



(b) Specific impulse.

Figure 42. - Performance levels for propulsion system 5 (EDMRJ).

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